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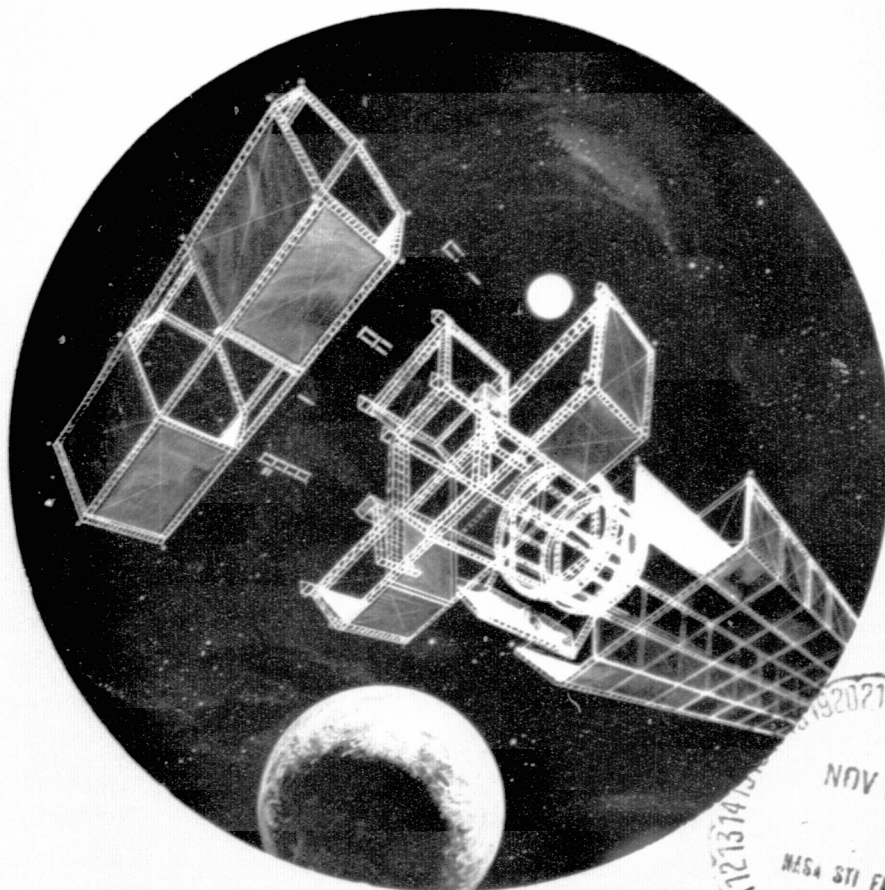
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(SPS) CONCEPT DEFINITION STUDY (EXHIBIT C)
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SD 78-AP-0115

**SATELLITE POWER SYSTEM (SPS)
CONCEPT DEFINITION STUDY
(EXHIBIT C)
MIDTERM PERFORMANCE REVIEW**

MARSHALL SPACE FLIGHT CENTER

SEPTEMBER 19-21, 1978



Rockwell International
Satellite Systems Division
Space Systems Group

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Second Quarterly Review

**SATELLITE POWER SYSTEMS (SPS)
CONCEPT DEFINITION STUDY
(EXHIBIT C)**

September 19-21, 1978

Prepared for:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER

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ALABAMA 35812

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Rockwell
International

BRIEFING OUTLINE

EXECUTIVE SUMMARY

G. M. HANLEY

REFERENCE CONCEPT CHARACTERISTICS

A. A. NUSSBERGER
R. E. OGLEVIE

REFERENCE CONCEPT CONSTRUCTION

R. E. SEXTON

MICROWAVE TRANSMISSION SYSTEM

G. D. O'CLOCK

TRANSPORTATION SYSTEM

R. P. BERGERON

EXPERIMENT/ VERIFICATION PROGRAM

W. R. RHOTE



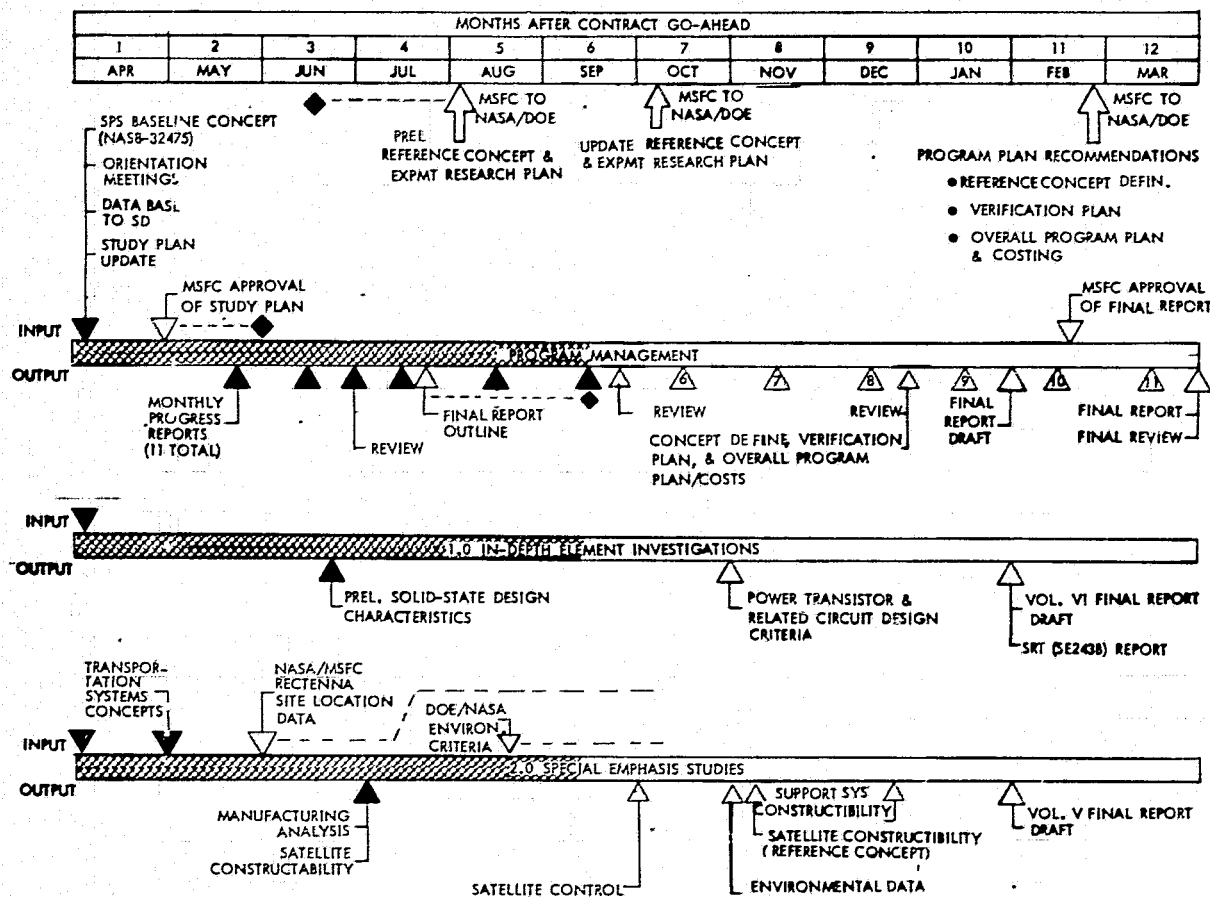
SPS CONCEPT DEFINITION STUDY SCHEDULE

During the first 3 months of the study, concept definition effort was concentrated on the MSFC/Rockwell baseline concept. Recently, NASA defined a coplanar satellite conceptual approach which has been the focus during this reporting period. This effort included several trade studies related to satellite design and also construction approaches for this satellite. A transportation system, consistent with this concept, also was studied, including an electric orbit transfer vehicle (EOTV) and a parallel-burn heavy lift launch vehicle (HLLV). These studies have lead to data that will be used to select a preferred satellite and transportation system approach as well as an approach for satellite construction.

Work on a solid-state microwave concept also continued and several alternative approaches were evaluated. Computer determination of an optimized transistor and circuit design continued.

Experiment/verification planning results in the development of a total solar array and microwave technology development plan as well as definition of near-term research to evaluate key technology issues.

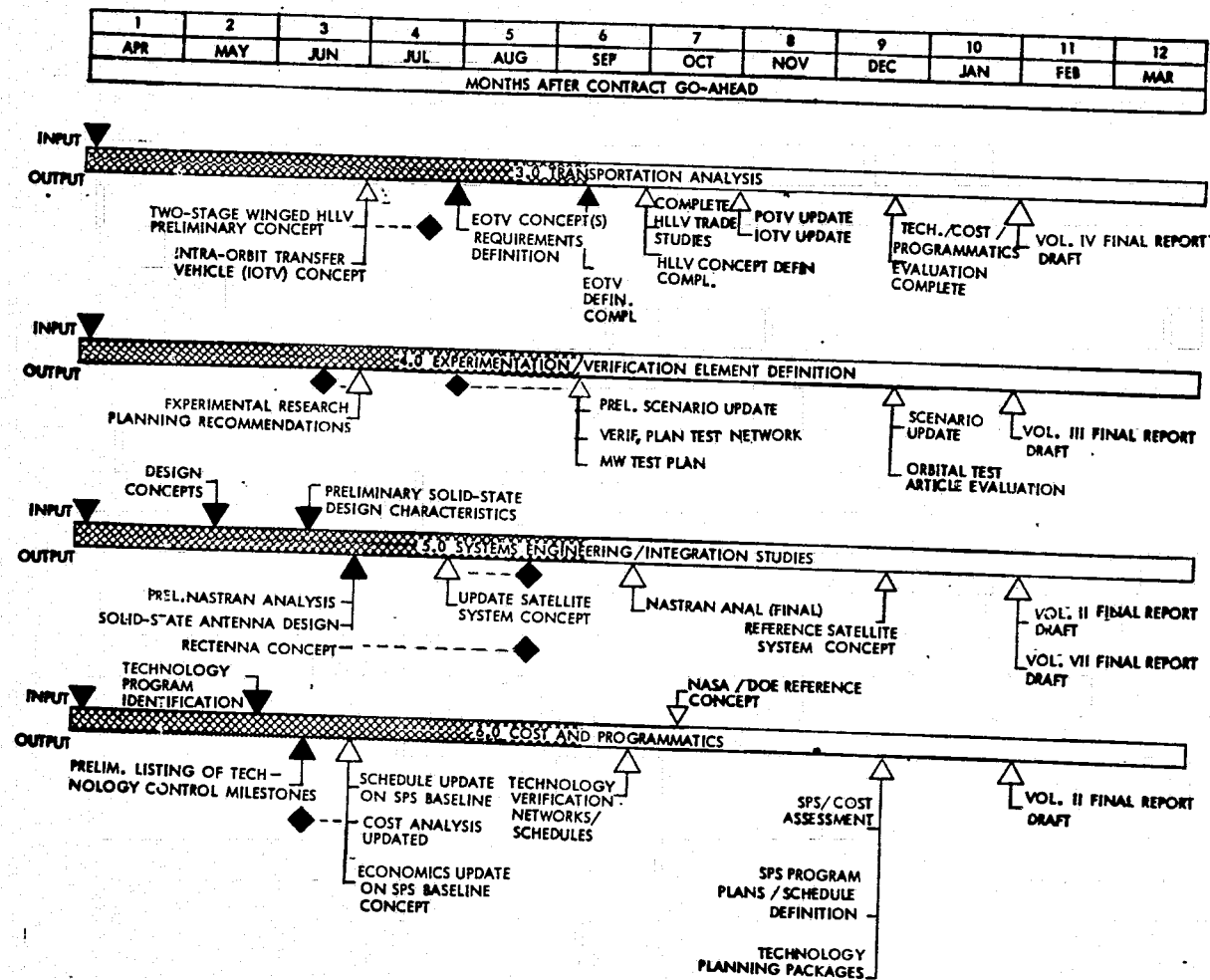
SPS CONCEPT DEFINITION STUDY SCHEDULE



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OF POOR QUALITY



SPS CONCEPT DEFINITION STUDY SCHEDULE (CONT.)



ORIGINAL PAGE 10
OF POOR QUALITY

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MAJOR AREAS OF PROGRESS

This chart is self-explanatory.

MAJOR AREAS OF PROGRESS

SYSTEM ENGINEERING

- ALTERNATIVES TO REFERENCE SATELLITE CONCEPT EVALUATED
- ALTERNATIVE SOLID STATE CONCEPTS EVALUATED
- NASTRANS STRUCTURAL ANALYSIS CONDUCTED

SPECIAL EMPHASIS STUDIES

- ALTERNATIVE CONSTRUCTION APPROACHES EVALUATED FOR REFERENCE SATELLITE AND ITS VARIATIONS

TRANSPORTATION STUDIES

- PARAMETRIC TWO-STAGE, PARALLEL-BURN VTO HLLV STUDY COMPLETED
- SILICON AND GaAs SOLAR ARRAY EOTV CONCEPTS DEVELOPED

PROGRAM PLANNING

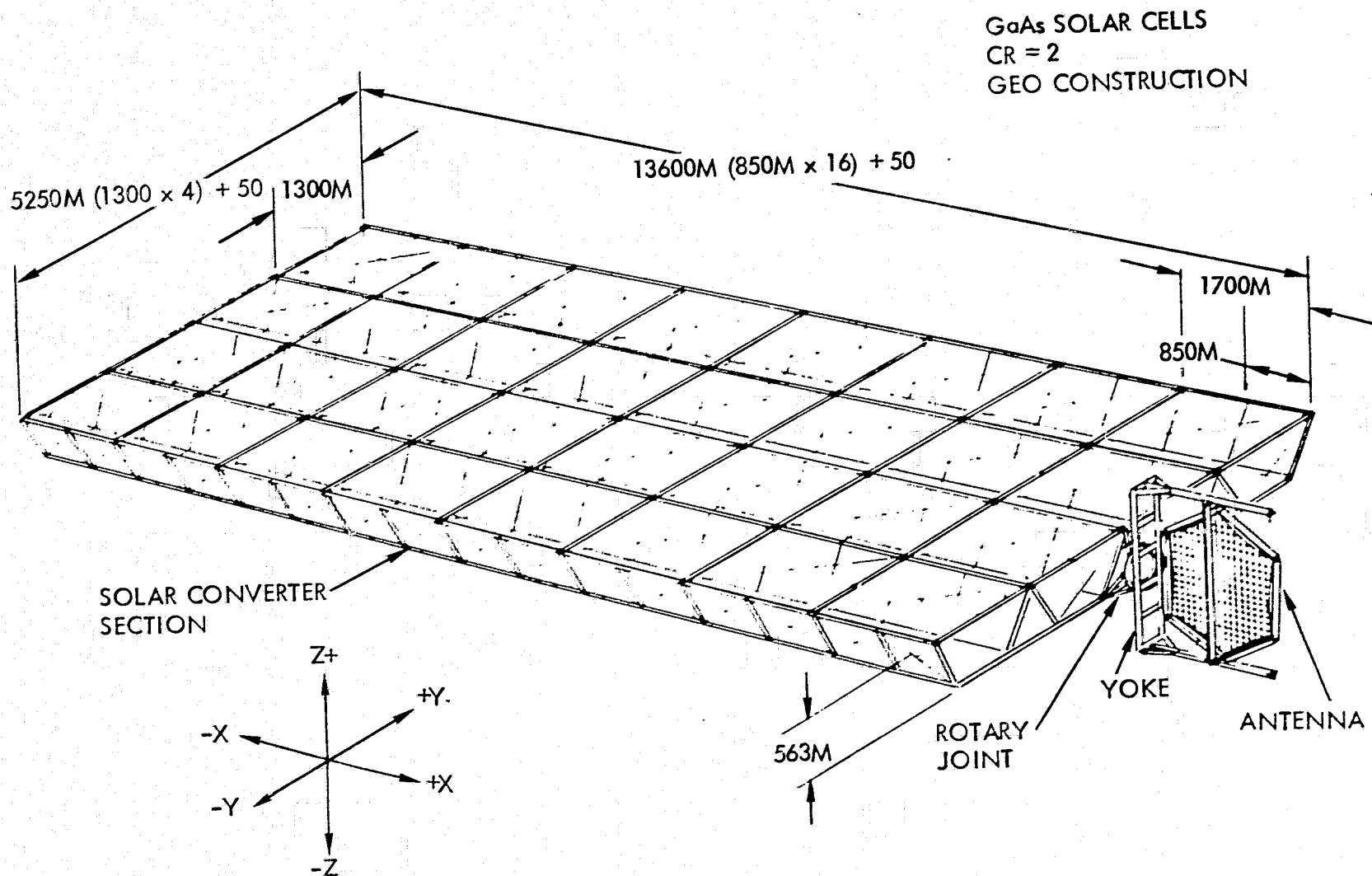
- EVOLUTIONARY PROGRAM APPROACH DEVELOPED
- IN-DEPTH SOLAR ARRAY AND MICROWAVE TECHNOLOGY DEVELOPMENT PROGRAMS IDENTIFIED



REFERENCE SATELLITE CONCEPT

This chart shows the reference satellite concept which was developed using the design characteristics shown on the next chart. The concept illustrated utilizes GaAlAs cells in a concentration ratio of 2 configuration. All four troughs are coplanar. The microwave antenna is located at the end of the configuration. This concept provides 5 GW of power at the utility interface on the ground.

REFERENCE SATELLITE CONCEPT



REFERENCE SATELLITE DESIGN CHARACTERISTICS

This chart is self-explanatory.

REFERENCE SATELLITE DESIGN CHARACTERISTICS

OVERALL DESCRIPTION

- 5-GW POWER TO UTILITY INTERFACE
- GEOSYNCHRONOUS CONSTRUCTION LOCATION
- SINGLE MICROWAVE ANTENNA
- GEOSYNCHRONOUS EQUATORIAL OPERATIONAL ORBIT

SUBSYSTEMS

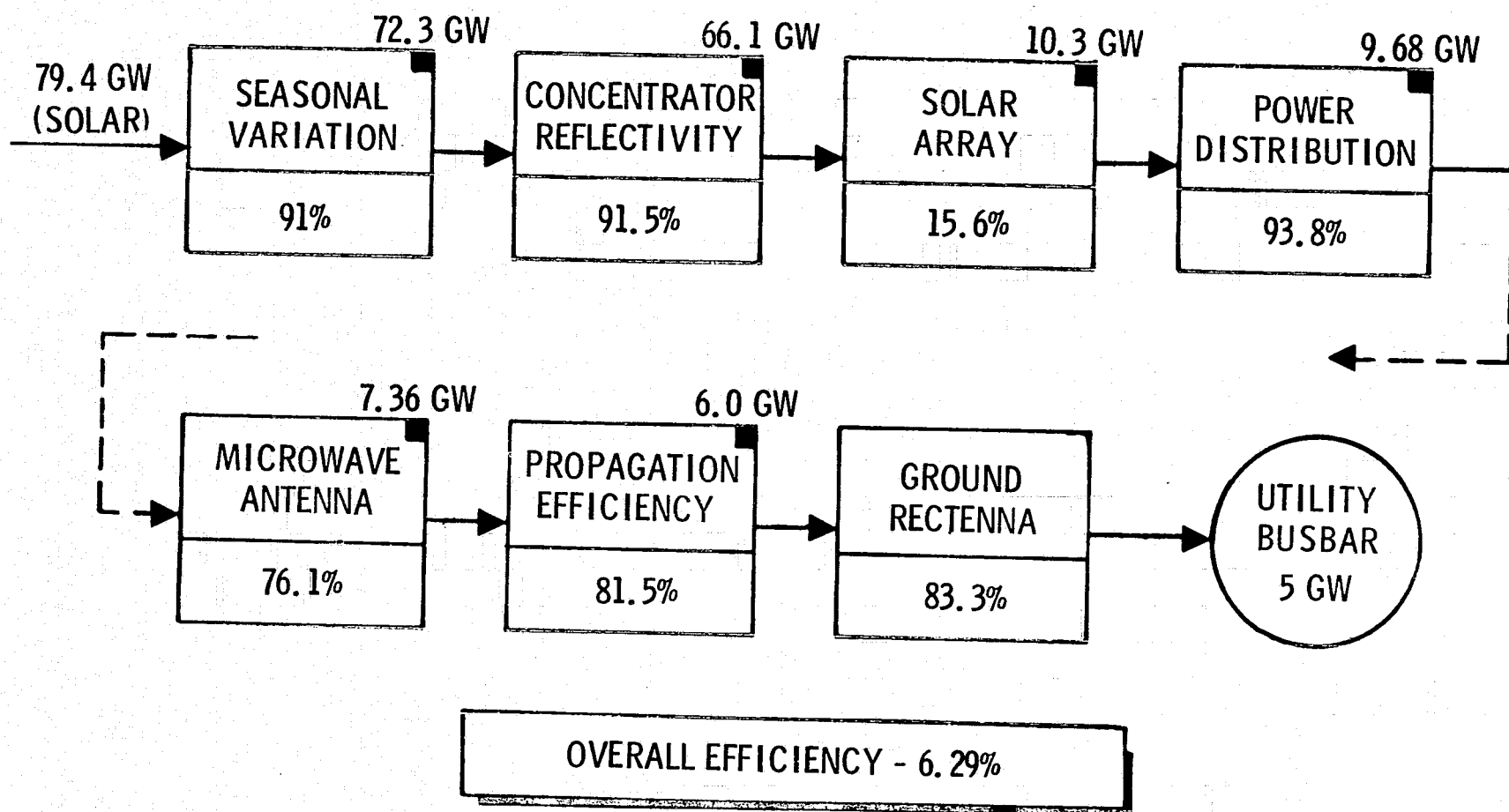
- POWER CONVERSION
 - GaAlAs SOLAR CELLS
- ATTITUDE CONTROL/STATIONKEEPING
 - Y-POP
- POWER DISTRIBUTION
 - 45.5 KV DC
- MICROWAVE ANTENNA
 - GAUSSIAN BEAM
 - 2.45-GHz FREQUENCY
 - ELECTRIC PHASE CONTROL
- STRUCTURE
 - COMPOSITES
 - BEAM MACHINE CONSTRUCTION
- INFORMATION MANAGEMENT
 - DISTRIBUTED
- CONCENTRATION RATIO = 2
- ARGON ION THRUSTERS
- STRUCTURE/WIRING NOT INTEGRATED
- RCR WAVEGUIDE PANELS



REFERENCE CONCEPT SYSTEM EFFICIENCY CHAIN

This chart is self-explanatory.

REFERENCE CONCEPT SYSTEM EFFICIENCY CHAIN



REFERENCE SATELLITE MASS STATEMENT

This chart presents the mass properties for the reference satellite shown on the previous chart. The total mass, 37.0×10^6 kg, includes a 25% growth factor. The dominant mass in the collector array is the solar array. Structural mass is only 10% of the total mass. Antenna mass is mainly comprised of the RF radiators and the klystrons.

REFERENCE SATELLITE MASS STATEMENT

SUBSYSTEM	WEIGHT (MILLION KG)
COLLECTOR ARRAY	
STRUCTURE & MECHANISMS	2.156
POWER SOURCE	8.709
POWER DISTRIBUTION & CONTROL	2.271
ATTITUDE CONTROL	0.116
INFORMATION MANAGEMENT & CONTROL	0.050
<u>TOTAL ARRAY (DRY)</u>	(13.302)
ANTENNA SECTION	
STRUCTURE & MECHANISMS	1.685
THERMAL CONTROL	2.457
MICROWAVE POWER	7.012
POWER DISTRIBUTION & CONTROL	4.516
INFORMATION MANAGEMENT & CONTROL	0.630
<u>TOTAL ANTENNA SECTION (DRY)</u>	(16.300)
TOTAL SPS DRY WEIGHT	29.602
GROWTH (25%)	7.400
TOTAL SPS DRY WEIGHT WITH GROWTH	37.002
PROPELLANT PER YEAR	0.085

ORIGINAL, IN FULL
OF POOR QUALITY



REFERENCE CONCEPT VARIATIONS AND IMPACTS

This chart lists the variations to the reference concept that were studied to provide data for selection of the best design and construction concept. It also indicates some of the most important impacts of the variations.

Both silicon and GaAs concepts were studied. The results indicate that the silicon concept is much heavier than the GaAs concept. The silicon concept is optimized without concentrators; whereas, the GaAs concept is near-optimum at $CR=2$. As the number of troughs vary from 3 to 6, the length-to-width of the planform decreases. The higher number of troughs generally have lower wiring weights but construction is more complex. The centrally-located antenna concepts have the lowest weights but are somewhat more complex to construct. For both centrally-located and end-located antenna concepts, three troughs appear to be the best compromise for weight and constructability.

REFERENCE CONCEPT VARIATIONS AND IMPACTS

VARIABLES	IMPACTS
CONCENTRATION RATIO ● CR = 1 ● CR = 2	● BEST FOR SILICON CONCEPT - SIMPLER CONSTRUCTION ● BEST FOR GaAs CONCEPT
NUMBER OF TROUGHS (L/W) ● 3, 4, 5, AND 6	● LOWER NUMBER FAVORED FOR CONSTRUCTION, HIGHER NUMBER FOR LOW CONDUCTOR WEIGHT
ANTENNA LOCATION ● CENTRAL ● END	● LOWEST WEIGHT, LARGE ROTATING JOINT ● INCREASED MASS (CONDUCTOR WEIGHT), SIMPLER CONSTRUCTION, REDUCED MICROWAVE ANTENNA COMPLEXITY (THERMAL)
SOLAR CELLS ● GaAs ● SILICON	● LOWEST WEIGHT AND COST CONCEPT



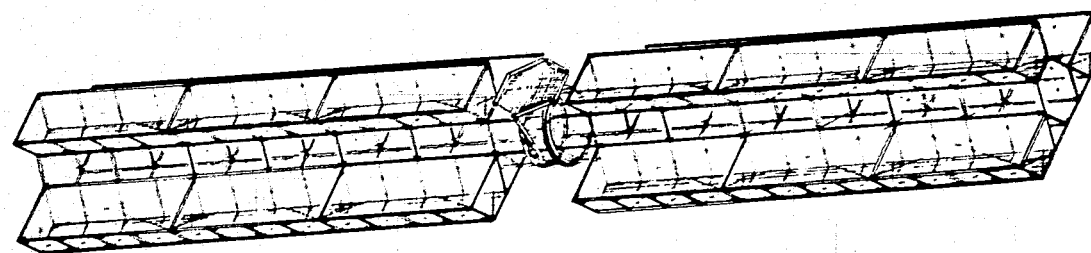
ANTENNA DESIGN CONCEPTS

This chart illustrates the matrix of antenna structure types (tension web and spaceframe) and antenna locations studied. Results of the trade studies indicates that the central antenna is preferred to reduce satellite mass. Previous studies considered cooling of the klystron collector from both the front and rear sides of the antenna. Radiation from the front side required a hole in the center of the antenna, thus reducing the propagation efficiency of the antenna. During this study, it was determined that it is thermally acceptable to radiate from the rear of the antenna, even with a centrally located antenna, provided that the slip rings are not located directly under the antenna. When comparing the two antenna structures, the thermal analysis concluded that the spaceframe antenna primary structure would encounter severe heating from the klystron collector (radiation from the rear of the antenna). High temperature composites are necessary to provide the needed structural strength. This is not a problem with the tension web concept.

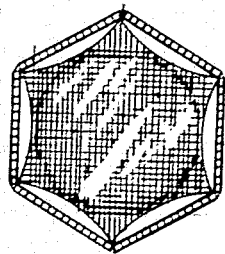
Studies of attitude control and stationkeeping indicated that little difference in total thrust and propellant requirement exists among the concepts. This is due to the dominance of the east-west stationkeeping caused by solar pressure. East-west stationkeeping is required to assure non-interference of an SPS with another SPS or another satellite. If east-west stationkeeping is constantly applied, all other attitude control and stationkeeping requirements can be obtained concurrently by combined ion-thruster applications. Additionally, some relief can be obtained from efficiency losses due to solar seasonal variations by inclining the satellite up to 9° toward the sun. This is another "free" benefit obtained concurrent with east-west stationkeeping.

ANTENNA DESIGN CONCEPTS

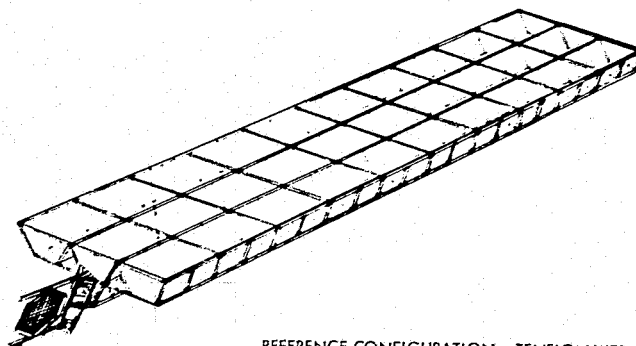
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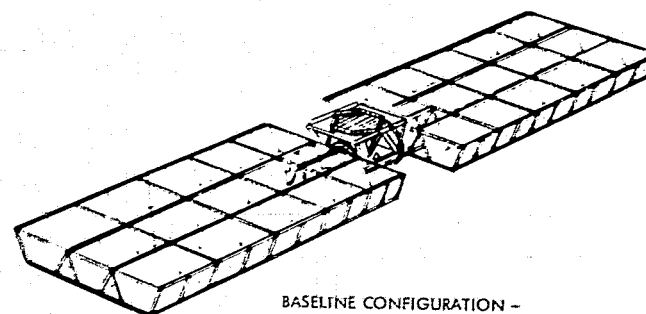
ROCKWELL BASELINE



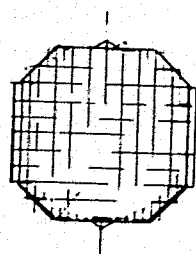
TENSION WEB



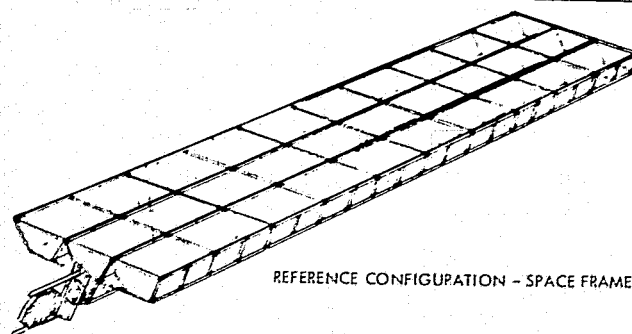
REFERENCE CONFIGURATION - TENSION WEB



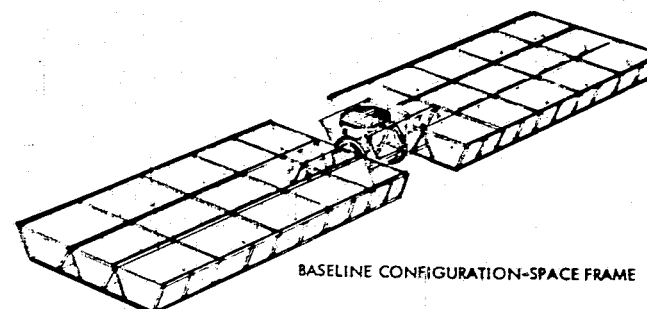
BASELINE CONFIGURATION -
SLIP RINGS NOT DIRECTLY OVER
ANTENNA



SPACEFRAME



REFERENCE CONFIGURATION - SPACE FRAME



BASELINE CONFIGURATION-SPACE FRAME

MAJOR SUBSYSTEM CONCLUSIONS

This chart is self-explanatory.

MAJOR SUBSYSTEM CONCLUSIONS

POWER DISTRIBUTION AND CONTROL

- CENTER -MOUNTED ANTENNA PREFERRED TO REDUCE SATELLITE MASS
- INCREASED NUMBER OF TROUGHS PREFERRED TO MINIMIZE POWER DISTRIBUTION MASS -- PARTICULARLY FOR END -MOUNTED ANTENNA CONCEPT

THERMAL/MICROWAVE ANTENNA

- CENTER -MOUNTED OR END -MOUNTED ANTENNAS ARE THERMALLY ACCEPTABLE EVEN WITH KLYSTRON COLLECTOR RADIATION OFF REAR OF ANTENNA
- SLIP RINGS SHOULD NOT BE LOCATED DIRECTLY UNDER ANTENNA
- SPACEFRAME ANTENNA PRIMARY STRUCTURE HAS SEVERE LOCAL HEATING FROM KLYSTRON COLLECTOR - REQUIRES HIGH TEMPERATURE COMPOSITES



MAJOR SUBSYSTEM CONCLUSIONS (CONTINUED)

ATTITUDE CONTROL/STATIONKEEPING

- EAST -WEST DRIFT DUE TO SOLAR P RESSURE REQUIRES CORRECTION
- IF CORRECTION IS CONTINUOUS, COMBINED ATTITUDE CONTROL AND STATIONKEEPING CORRECTIONS RESULT IN SAME PROPELLANT REQUIREMENT AS STATIONKEEPING ALONE FOR ALL CONCEPTS STUDIED

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MASS PROPERTIES COMPARISONS

This chart shows the mass properties for GaAs (CR=2) concepts with end- and center-mounted antennas and for a silicon concept with an end-mounted antenna. The antenna section has the same mass for all concepts. The silicon concept is much heavier (by 20×10^6 kg) than the GaAs concept, primarily due to the solar cell mass. The center-located antenna is the lowest weight concept (by 2×10^6 kg) primarily due to power distribution and control mass.

MASS PROPERTIES COMPARISONS ~ 10⁶ KG

SUBSYSTEM	GaAs CR = 2		SILICON CR = 1
	(END ANTENNA)	(CENTER ANTENNA)	(END ANTENNA)
<u>COLLECTOR ARRAY</u>			
STRUCTURE & MECHANISMS	1.811	1.561	1.607
POWER SOURCE	8.44	8.44	23.988
POWER DIST. & CONTROL	2.889	1.503	3.60
ATTITUDE CONTROL	.116	.116	.116
INFORMATION MGMT & CONTROL	.050	.050	.050
<u>TOTAL ARRAY (DRY)</u>	13.306	11.670	29.361
<u>ANTENNA SECTION</u>			
STRUCTURE & MECHANISMS	1.486	1.486	1.486
THERMAL CONTROL	2.457	2.457	2.457
MICROWAVE POWER	7.012	7.012	7.012
POWER DISTRIB. & CONTROL	4.516	4.516	4.516
INFORMATION MGMT & CONTROL	.630	.630	.630
<u>TOTAL ANTENNA SECTION (PAY)</u>	16.101	16.101	16.101
<u>TOTAL SPS DRY WEIGHT</u>	29.407	27.771	45.462
GROWTH (25%)	7.352	6.943	11.366
<u>TOTAL SPS DRY WT. WITH GROWTH</u>	36.759	34.714	56.828
PROPELLANT PER YEAR			

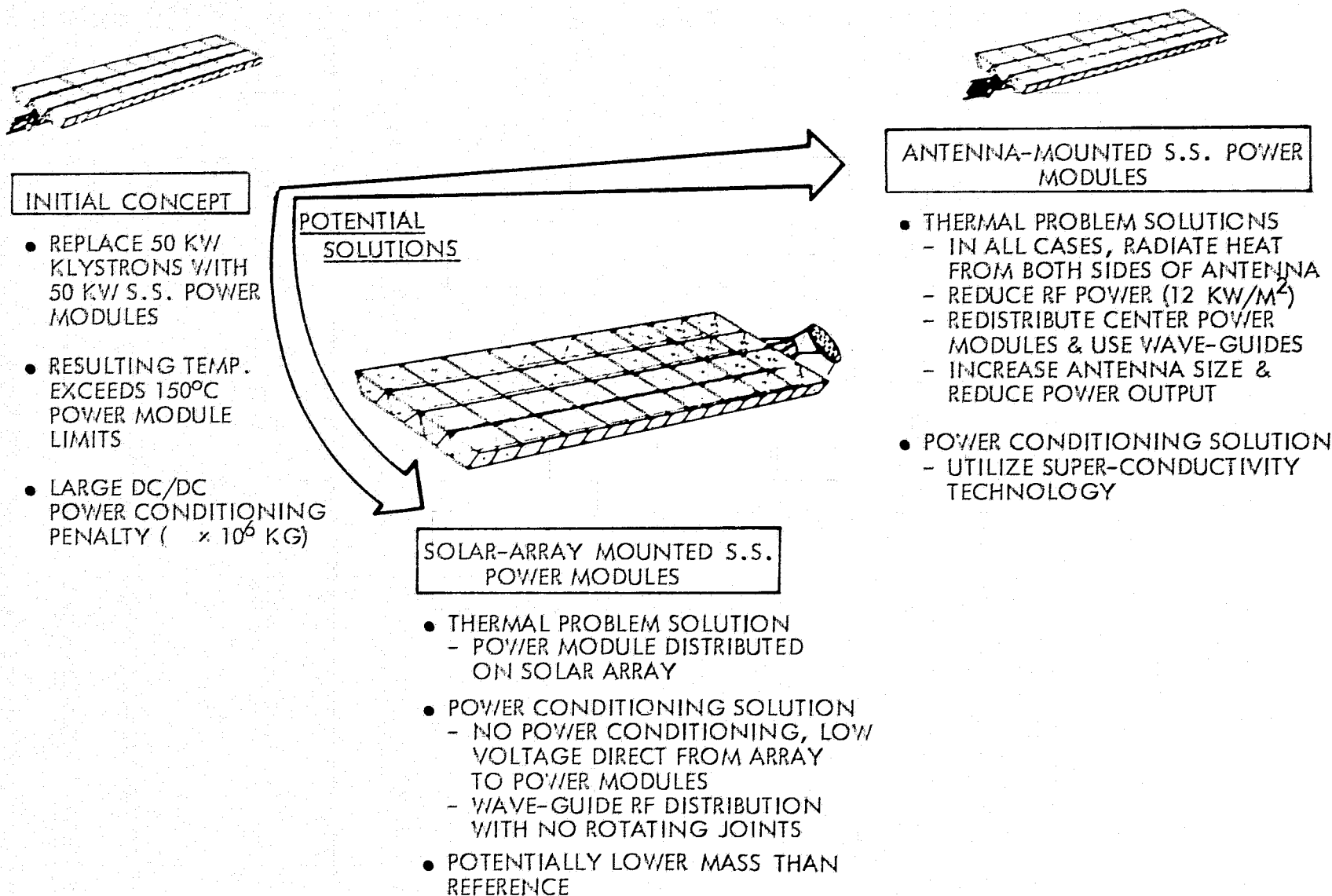
ALTERNATIVE SOLID-STATE CONCEPTS

The concept initially studied for application of solid-state technology assumed that 50 kW solid-state power modules replaced the 50 kW klystrons in an antenna configuration basically the same as that used for the klystrons. In order to provide adequate efficiency, the temperature of the solid-state devices needs to be maintained at 150°C. For this reason, it was determined that the current antenna configuration could not be used. Additionally, the low voltage required by the solid-state modules resulted in a large DC/DC power conditioning penalty. Two basic approaches were studied to alleviate the problems described above: one of these still maintained the antenna-mounted power modules while the other had the power modules located on the solar array.

For the antenna-mounted concepts, it is necessary to radiate thermal energy from both sides of the antenna if reasonable antenna sizes are to be maintained. Alternative approaches with two-sided radiation include: (1) a reduction of RF power in the center of the antenna to 21 kW/m², (2) redistribution of the center power modules to increase their spacing and the use of wave guides to direct RF energy to the appropriate microwave radiators, and (3) increase antenna diameter to increase thermal and microwave radiation area while simultaneously reducing power output from the satellite. A potential solution to the large masses due to step-down from high to low voltage is to conduct the power required by the antenna at low voltage using superconductivity technology to reduce conductor weights.

Another approach to solving the thermal problem as well as the power conditioning problem is to locate the solid-state power modules on the solar array. Power is provided directly from the array at the required voltage. Wave guides are used to direct the RF energy to a horn located on the end of the array that feeds a lens antenna. This concept has no rotating wave guide joints. This concept appears to have the potential for lower mass than the reference concept. However, it does introduce new technology problems related to high power wave guides.

ALTERNATIVE SOLID STATE ANTENNA CONCEPTS



COMPARISON OF CONSTRUCTION CONCEPT APPROACHES

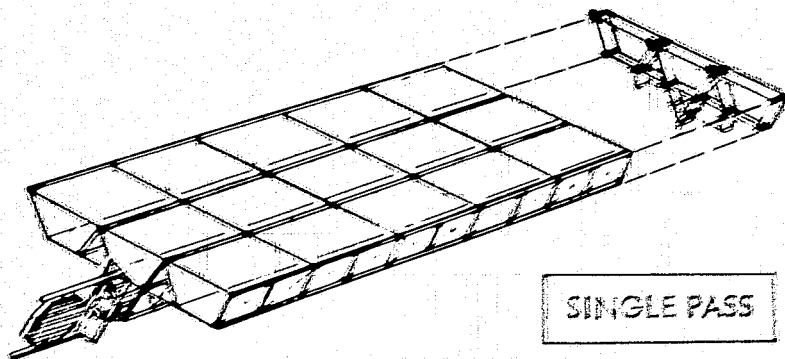
A comparison of construction concepts for the three trough satellite configuration is shown. The single pass concept is shown on the left; the serpentine concept on the right. Program ground rules specify 180 days for satellite construction. The serpentine concept, entailing sequential construction of the troughs, requires the total allocated time. For the single pass concept, the fabrication time can be reduced to 108 days, if desired.

Since, in the single pass concept, all troughs are constructed simultaneously, a larger crew is required than for the serpentine. However, if the entire 180 days is utilized, the crew size would be reduced.

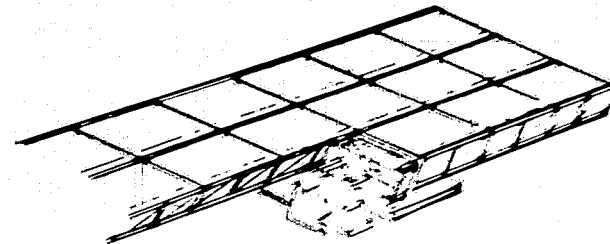
The masses of the two construction facilities is comparable. However, the serpentine concept, because of the complexities of the sliding platform and facility, and the number of complex operations required to construct one satellite, has been accorded a higher complexity, or risk factor.

The number of tribeam fabricators required for each concept is comparable; however, it is noted that duplicate sets are installed in the serpentine facility because each of the two sides, or faces, is used during the construction cycle.

COMPARISON OF CONSTRUCTION CONCEPT APPROACHES



SINGLE PASS



MULTI-PASS
SERPENTINE

• CONSTRUCTION TIME	108 DAYS
• CREW SIZE (AVERAGE)	376
• CONST. FACILITY MASS (KG)	5.3×10^6
• RELATIVE CONST. COMPLEXITY	1
• TRIBEAM FABRICATORS	28

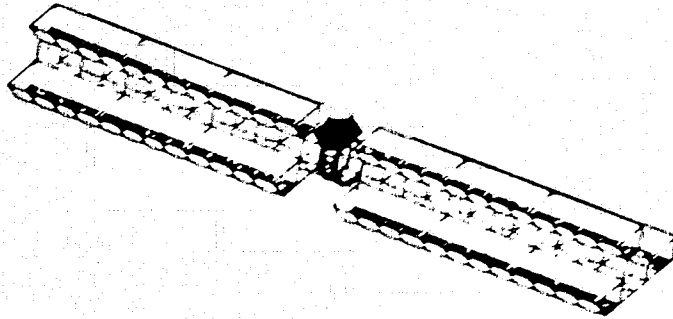
180 DAYS
266
5.2×10^6
4.5
26

NASTRANS STRUCTURAL ANALYSIS SUMMARY

This chart summarizes preliminary results of a NASTRAN structural analysis of the three-trough Rockwell concept. Both composite and aluminum structure were analyzed. Thermal constraints include equilibrium operating temperatures, a construction temperature of 0°C and a temperature during the solar eclipse by the earth of -150°C .

The results indicated maximum deflections of the aluminum array structure of 100 m compared to only 1 m for the composite structure. Rotating joint deflections of 0.88 m were obtained compared with negligible composite deflections. Despite these large deflections, it was concluded that an aluminum structure could be built to SPS requirements. However, the results indicated that a structure material thickness of up to 30 mils is necessary compared to 10 mils previously assumed. This would result in a structural weight up to 10×10^6 kg for aluminum compared to 1.2×10^6 kg for a composite structure.

NASTRANS STRUCTURAL ANALYSIS SUMMARY



ASSUMPTIONS

- 3-TROUGH CONFIGURATION
- THERMAL CONSTRAINTS
 - EQUILIBRIUM OPERATING TEMPERATURES
 - MINIMUM TEMPERATURE DURING ECLIPSE = -150°C
 - CONSTRUCTION TEMPERATURE = 0°C
- COMPOSITE & ALUMINUM STRUCTURES

RESULTS

- MAXIMUM DEFLECTIONS
 - ALUMINUM - 100 M
 - COMPOSITE - 1.1 M
- ROTATING JOINT DEFLECTIONS
 - ALUMINUM - 0.88 M
 - COMPOSITE - NEGLIGIBLE
- ADDITIONAL STRUCTURAL ANALYSIS INDICATES THAT ALUMINUM STRUCTURE THICKNESS SHOULD BE 30 MILS
 - RESULTS IN UP TO 10.0 MILLION KG ALUMINUM STRUCTURE WEIGHT COMPARED TO COMPOSITE WEIGHT OF 1.2 MILLION KG
- DESPITE LARGE DEFLECTIONS, ALUMINUM STRUCTURE WILL WORK
- COMPOSITE PROBLEM IS 30 YEAR LIFETIME



VTO/HL HLLV CONCEPT TRADE STUDIES

This chart is self-explanatory.

VTO/HL HLLV CONCEPT TRADE STUDIES

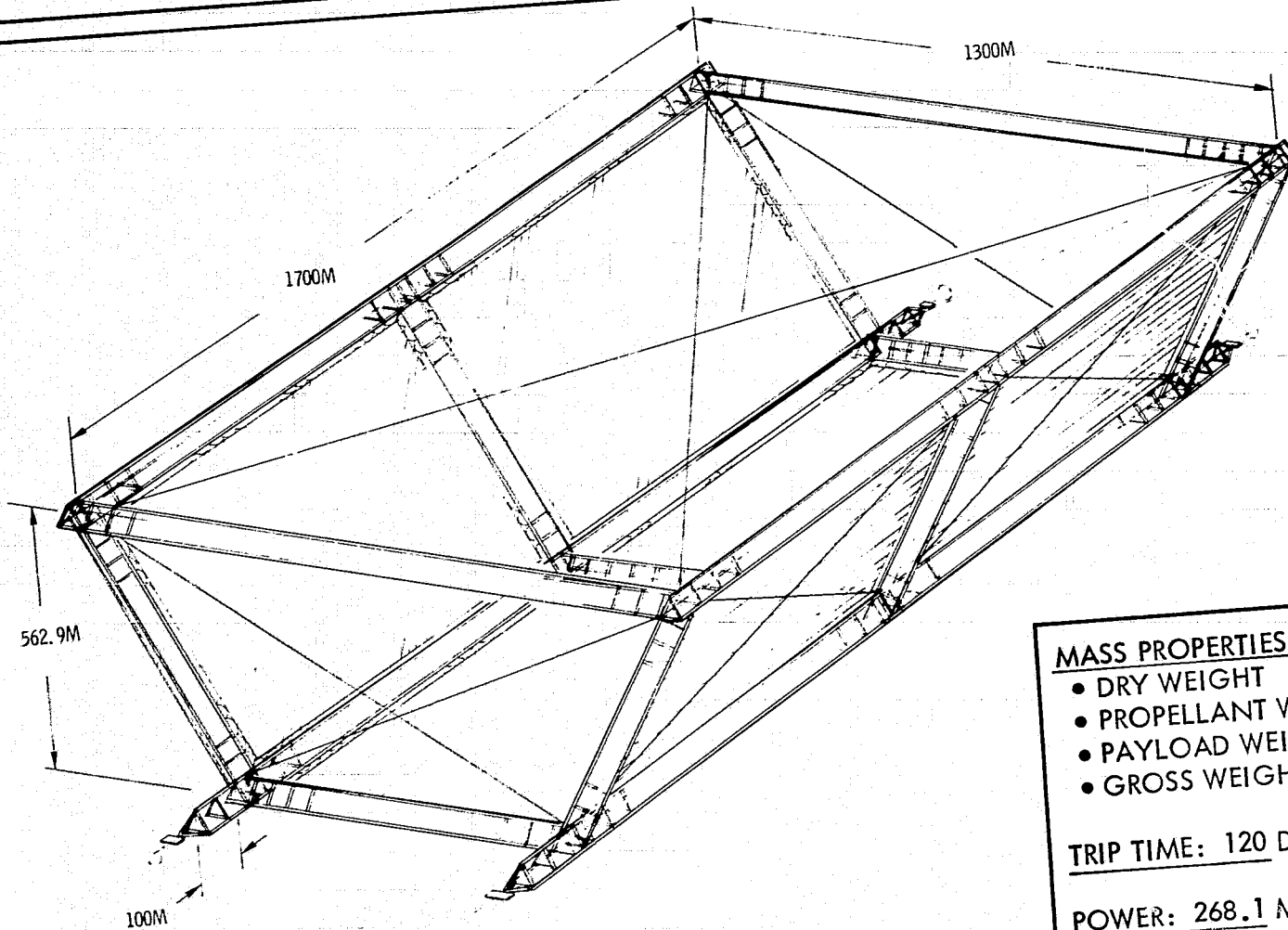
- PARALLEL VERSUS SERIES BURN
- FIRST STAGE FUEL OPTIONS - LH_2 , CH_4 , RP
- LIFT-OFF THRUST -TO -WEIGHT
- STAGING VELOCITY
- DRY WING VERSUS WET WING



EOTV CONCEPT CHARACTERISTICS

The selected EOTV concept utilizes two SPS solar array bays, and four gimbaled thruster module arrays. Each thruster array contains twenty thrusters, however, only sixty-four thrusters are capable of firing simultaneously (the thrust indicated is for sixty-four thrusters). Four electric thrusters utilizing stored electrical power are also used for attitude hold only during occultation periods. The net power from the solar array to the thrusters includes line losses of 6% and lifetime degradation of 15% from the available solar array power of 335.5 megawatts.

EOTV CONCEPT CHARACTERISTICS



20 (76 CM) THRUSTERS
AT EACH CORNER

ORIGINAL PROJECT
OF POOR QUALITY

- BUILT BY SATELLITE CONSTRUCTION FACILITY
- POTENTIALLY SPS DEMONSTRATION ARTICLE WHEN MODIFIED

MASS PROPERTIES (10^6 KG)

• DRY WEIGHT	0.809
• PROPELLANT WEIGHT	0.547
• PAYLOAD WEIGHT	5.311
• GROSS WEIGHT	6.667

TRIP TIME: 120 DAYS UP & 12.5 DAYS RETURN

POWER: 268.1 MW (NET FROM ARRAY)

THRUST: 4462 NEWTONS



PROGRAM PLANNING EMPHASIS

This chart is self-explanatory.

PROGRAM PLANNING EMPHASIS

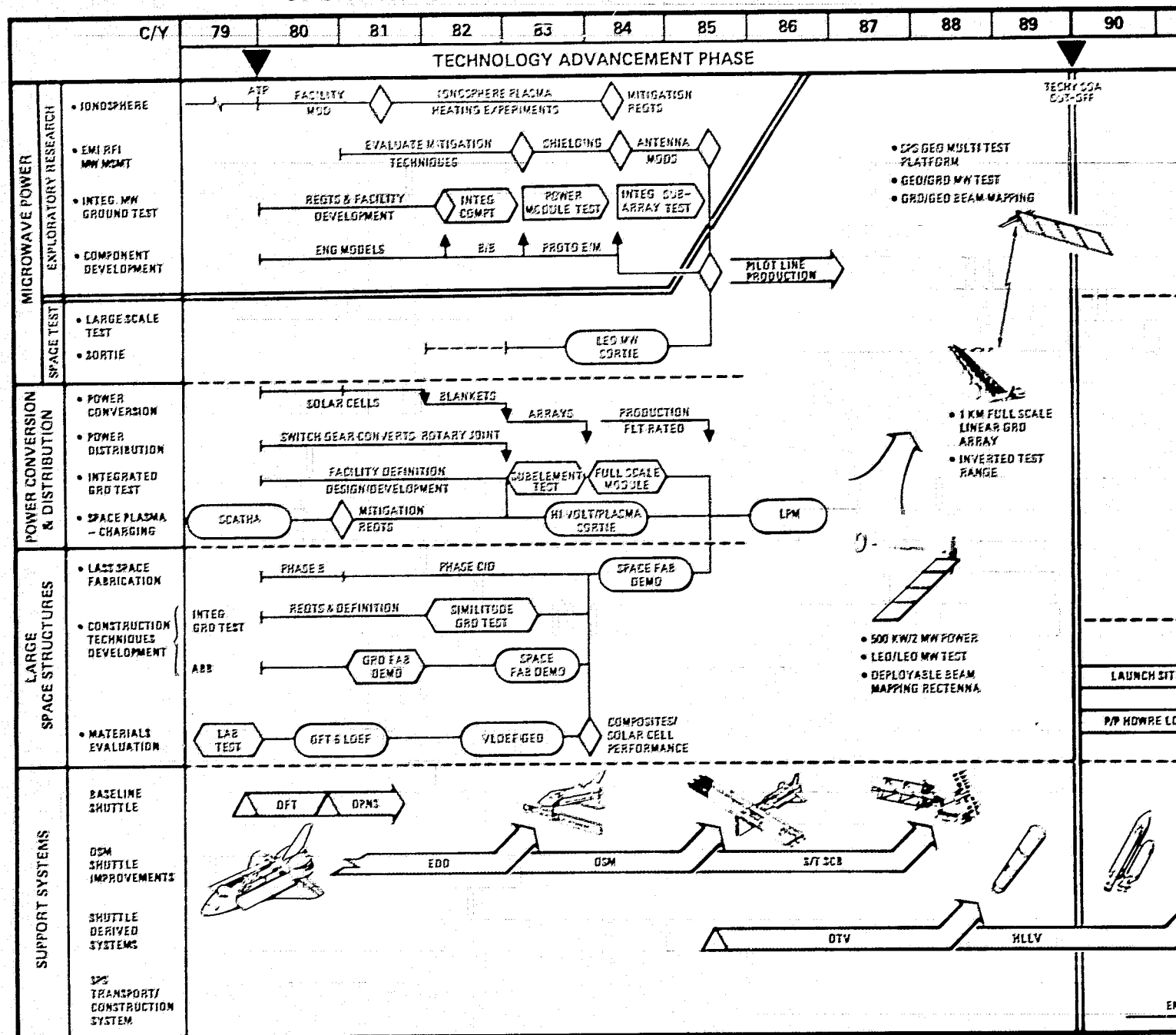
- OVERALL PROGRAM EVOLUTION
 - INTERACTIONS OF EOTV, CONCEPT DEMONSTRATOR, AND FACILITIES BUILD-UP
 - EARLY SPACE TEST ARTICLE IMPLICATIONS
- IN-DEPTH TECHNOLOGY PLANNING
 - SOLAR BLANKET DEVELOPMENT PROGRAM
 - MICROWAVE SYSTEM DEVELOPMENT PROGRAM
- FUTURE EFFORT SHOULD FOCUS ON POWER DISTRIBUTION AND CONTROL



SPS TECHNOLOGY ADVANCEMENT PLAN

The exploratory research plan segment is contained within the double lines and provides the "seed-bed" for prototype MPTS development. Planned and on-going NASA enabling technology for power conversion/distribution and large space structures development are structured to provide evolutionary timelines leading to large SPS-type subscale space test articles during the end of the decade. The commitment to large-scale SPS development ground and space test activity will come about 1985 based upon Exploratory Research Program results and will require funding levels on the order of several billions of dollars.

SPS TECHNOLOGY ADVANCEMENT PLAN



- CS/GES MULTI TEST PLATFORM
- GEO/GRD MW TEST
- GRD/GE0 BEAM MAPPING

- 1 KM FULL SCALE LINEAR GRD ARRAY
- INVERTED TEST RANGE

- 500 KW/2 MW POWER
- LEO/LEO MW TEST
- DEPLOYABLE BEAM MAPPING RECTENNA

ORIGINAL, IT
OF POOR QUALITY

CONCLUSIONS

This chart is self-explanatory.

CONCLUSIONS

ALTERNATIVE REFERENCE SATELLITE CONCEPTS

- CR=1 PREFERRED FOR SILICON CONCEPT & CR=2 PREFERRED FOR GaAs CONCEPT
- COPLANAR CONCEPTS COMPARABLE IN MASS TO ROCKWELL-DEVELOPED POINT DESIGN
- CENTER-MOUNTED ANTENNA CONCEPTS HAVE LOWER MASS, MORE COMPLEX CONSTRUCTION
- END-MOUNTED ANTENNA CONCEPTS HAVE ATTITUDE CONTROL REQUIREMENTS SIMILAR TO CENTER-MOUNTED CONCEPTS
- THREE TROUGHS PREFERRED FOR CENTER-MOUNTED ANTENNA CONCEPTS & FOUR TROUGHS PREFERRED FOR END-MOUNTED ANTENNA CONCEPTS
- SINGLE-PASS SATELLITE CONSTRUCTION PREFERRED

SOLID STATE CONCEPTS

- ANTENNA-MOUNTED POWER MODULE CONCEPTS REQUIRE ADVANCED-TECHNOLOGY SUPERCONDUCTIVITY FOR ACCEPTABLE POWER CONDITIONING MASS
- SOLAR-ARRAY-MOUNTED POWER MODULE CONCEPTS MAY HAVE LOWER MASS THAN CURRENT REFERENCE CONCEPT-ADVANCED-TECHNOLOGY WAVE-GUIDE RF DISTRIBUTION REQUIRED

CONCLUSIONS (CONTINUED)

NASTRANS ANALYSIS

- EITHER ALUMINUM OR COMPOSITE STRUCTURE WORKABLE
- INCREASE OF ALUMINUM STRUCTURAL MATERIAL THICKNESS UP TO 30 MILS MAY SIGNIFICANTLY INCREASE SATELLITE MASS

TRANSPORTATION STUDIES

- PREFERRED CONCEPT HAS LOX/RP FIRST STAGE AND LOX/LH₂ SECOND STAGE WITH PROPELLANT CROSS-FEED
- FIRST AND SECOND STAGES NEAR -EQUAL IN VOLUME

PROGRAM PLANNING

- EARLY GaAs SOLAR CELL FEASIBILITY DEMONSTRATION CRITICAL TO SPS 1980 PROGRAM DECISION - DEMONSTRATION PROGRAM PROPOSED TO DOE

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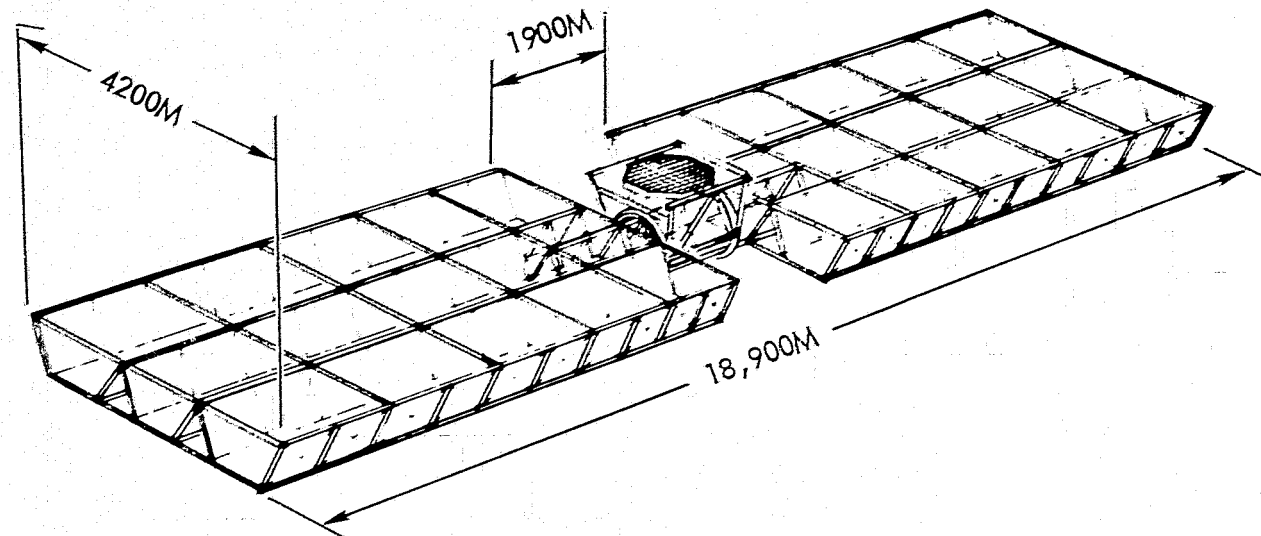
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RECOMMENDED SATELLITE CONCEPT

As a result of the trade studies on the reference concept, the concept illustrated on this chart is recommended for point design study. This coplanar concept has a single microwave antenna located in the center of the solar array. The solar array has 3 troughs containing GaAlAs solar cells with flat reflectors that give $CR=2$. Construction and operation of this satellite are at geosynchronous orbit.

RECOMMENDED SATELLITE CONCEPT



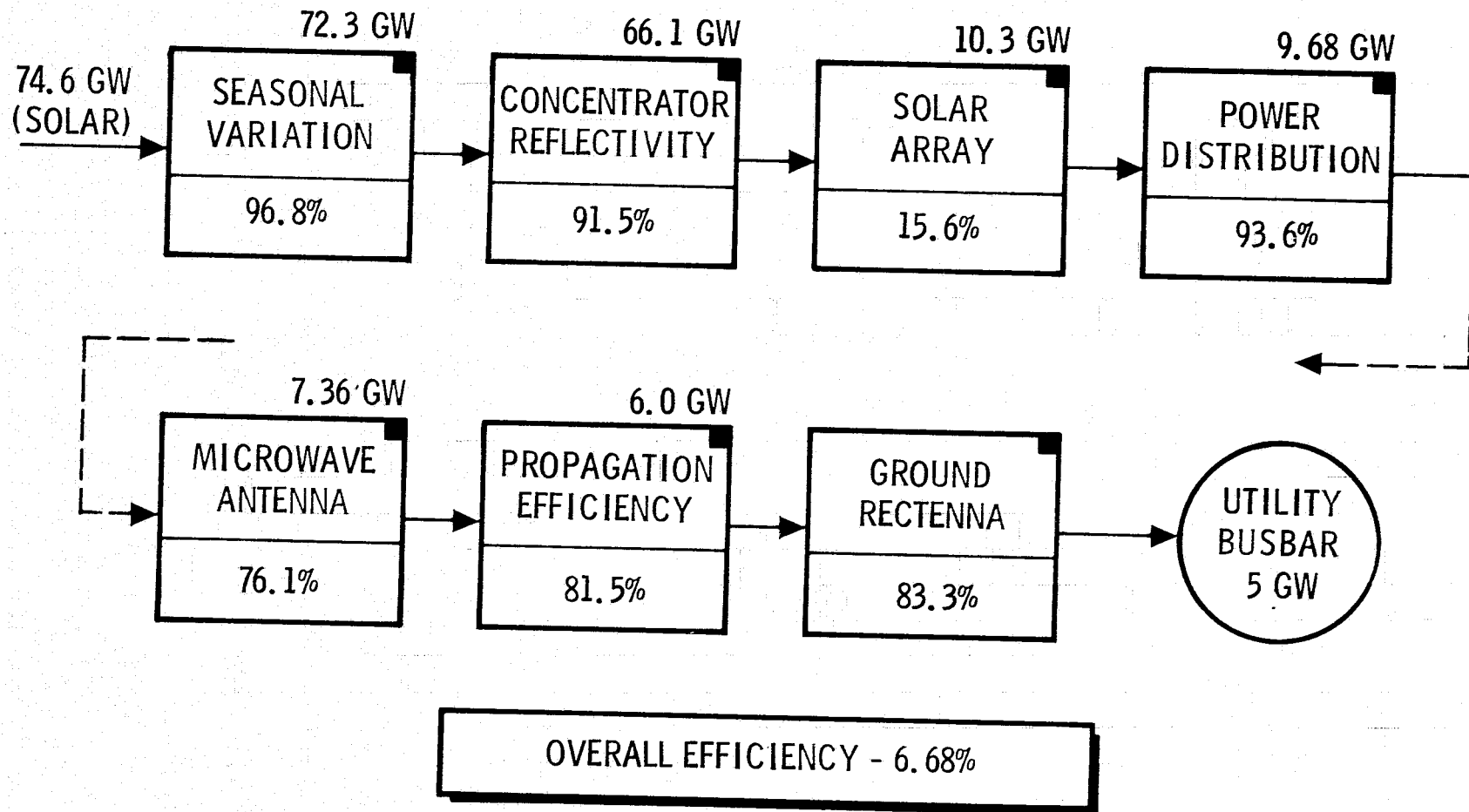
- 5 GW AT UTILITY INTERFACE
- CENTER-MOUNTED ANTENNA
- 3 TROUGH, COPLANAR ARRAY
- CR = 2, GaAlAs CELLS
- GEOSYNCHRONOUS ORBIT CONSTRUCTION & OPERATION

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EOL EFFICIENCY CHAIN FOR RECOMMENDED CONCEPT

This chart is self-explanatory.

EOL EFFICIENCY CHAIN FOR RECOMMENDED CONCEPT



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FUTURE EFFORT

This chart is self-explanatory.

FUTURE EFFORT

- SELECT SYSTEM CONCEPT AND CONDUCT POINT DESIGN
- COMPLETE SATELLITE AND SUPPORT SYSTEM CONSTRUCTABILITY ANALYSIS
- DETERMINE STATUS AND FEASIBILITY OF SUPERCONDUCTIVITY AND HIGH-POWER WAVE-GUIDE TECHNOLOGIES
- COMPLETE SOLID-STATE TRANSISTOR AND CIRCUIT DESIGN
- COMPLETE NASTRANS ANALYSIS
- COMPLETE HLLV CONCEPT DEFINITION AND UPDATE IOTV AND POTV DEFINITION
- COMPLETE DEFINITION OF GaAs SOLAR CELL DEMONSTRATION PROGRAM
- UPDATE PROGRAM DEFINITION AND COSTS



BRIEFING OUTLINE

EXECUTIVE SUMMARY

G. M. HANLEY

REFERENCE CONCEPT CHARACTERISTICS

A. A. NUSSBERGER
R. E. OGLEVIE

REFERENCE CONCEPT CONSTRUCTION

R. E. SEXTON

MICROWAVE TRANSMISSION SYSTEM

G. D. O'CLOCK

TRANSPORTATION SYSTEM

R. P. BERGERON

EXPERIMENT/ VERIFICATION PROGRAM

W. R. RHOTE



TASK 5 SYSTEM ENGINEERING/INTEGRATION STUDY LOGIC

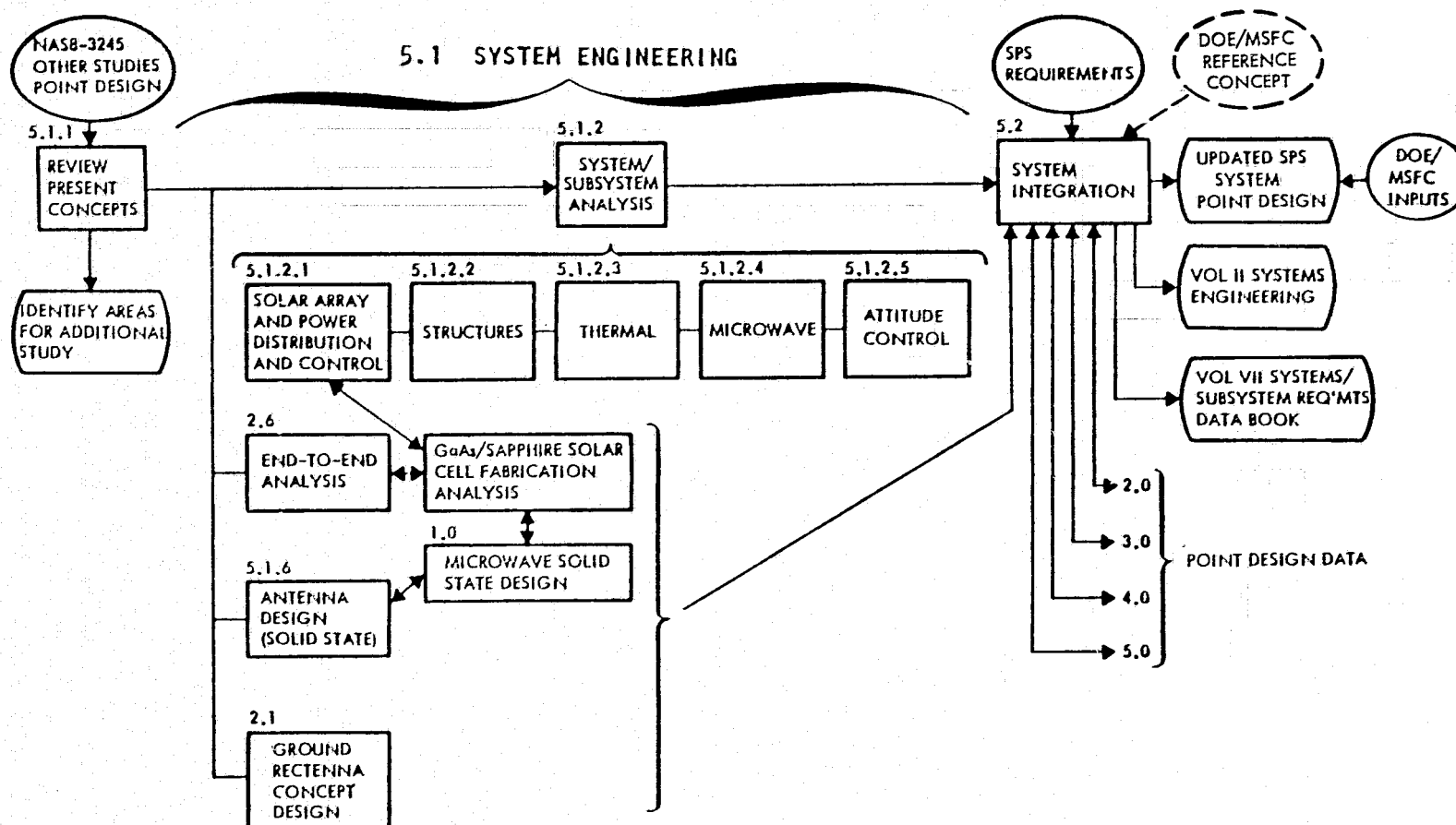
In Task 5, the data generated throughout the study and DOE/MSFC inputs are used to update the present SPS design concept. The logic diagram is shown. Major subtasks are:

- 5.1 SPS Systems Engineering (Space and Ground Elements)
- 5.2 SPS Systems Integration (Space and Ground Elements)

System Engineering - This subtask evaluates the photovoltaic design concept on the basis of new data to improve the design configuration, and to resolve or reduce key areas of uncertainty in the concept. This past quarters effort consisted of documenting the updated SPS system point design which resulted from previous study effort (e.g., the two tier concept) and system/subsystem analysis and trade-offs on the coplanar reference concept. These trade-offs are an integral part of the narrowing down process in configuration selection for point design definition.

System Integration - This subtask is a major program activity necessary to establish a final SPS system point design definition. The major output of this program activity is a set of summary requirements/point design documents describing the physical and performance characteristics of each of the elements making up the satellite system, and a detailed definition of its interfaces with transportation and support systems.

TASK 5 SYSTEM ENGINEERING/INTEGRATION STUDY LOGIC

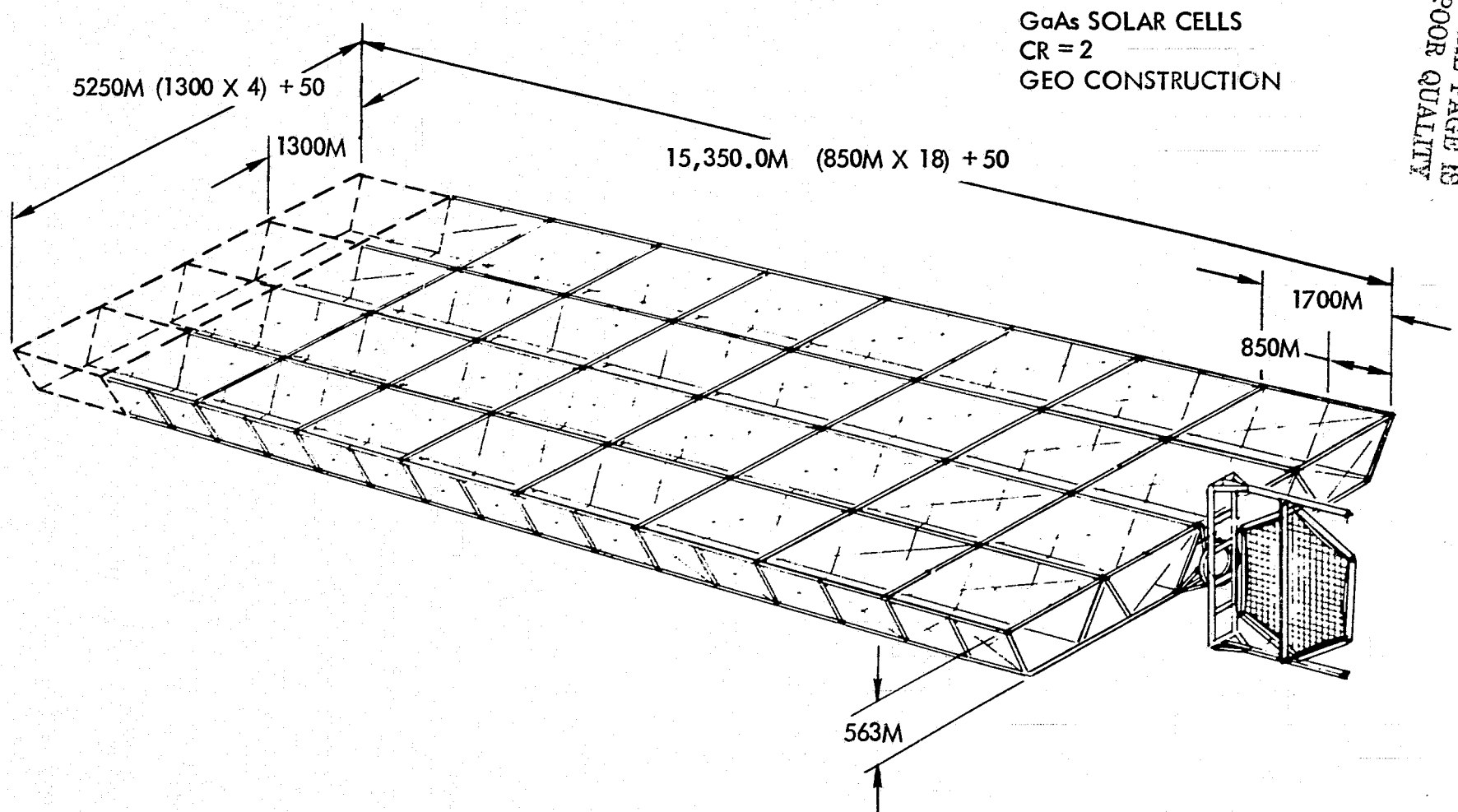


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COPLANAR SATELLITE CONFIGURATION AND CONSTRUCTION CONCEPT

This chart shows the reference coplanar satellite configuration used in the various trade off comparisons. This concept utilizes GaAlAs solar arrays at a concentration ratio of 2. The solar panels are arranged in a 4x9 matrix to provide power to the end mounted antenna. The last bay of solar panels can be removed by reducing the misorientation allowance during summer solstice from 23.5° to 14.48° by tipping the entire configuration 9.02°. This approach is covered within the attitude control and stationkeeping subsystem considerations and has a major impact on solar array sizing, i.e., reducing the overall solar array area requirements from $30.6 \times 10^6 \text{ m}^2$ to $28.4 \times 10^6 \text{ m}^2$.

COPLANAR SATELLITE CONFIGURATION & CONSTRUCTION CONCEPT



REPRESENTATIVE TROUGH CONFIGURATION

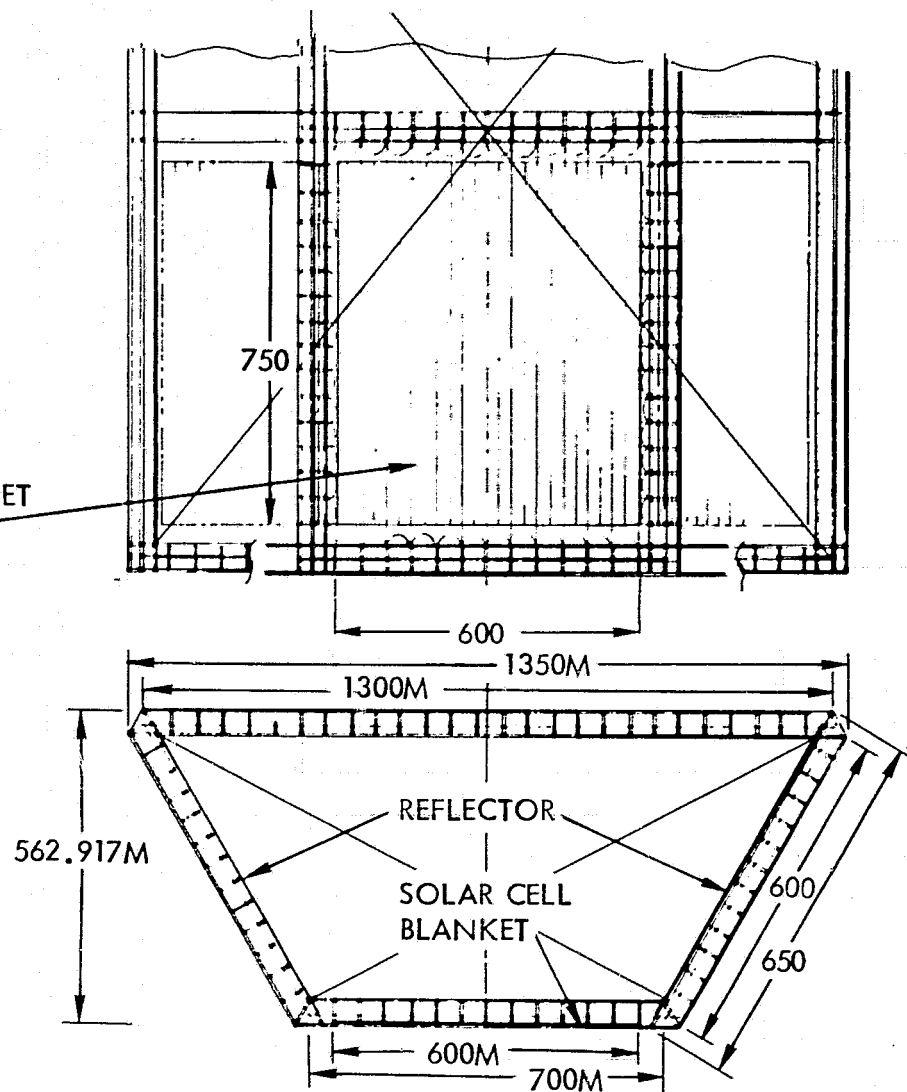
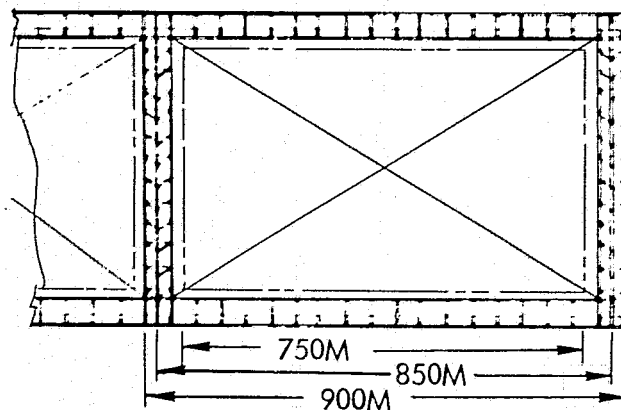
The solar array/blanket cross section is shown. A basic module with dimensions of 600 m wide $\times$ 750 m length is utilized in the design. Two of these modules are combined to provide the necessary 45.5 kV to the power distribution network.

REPRESENTATIVE TROUGH CONFIGURATION

GaAs SOLAR CELLS
CR = 2
GEO CONSTRUCTION

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24 SOLAR BLANKET
STRIPS/BAY



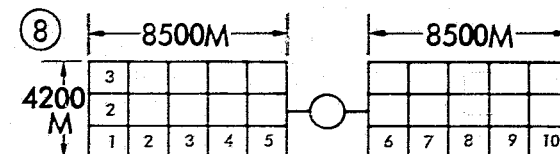
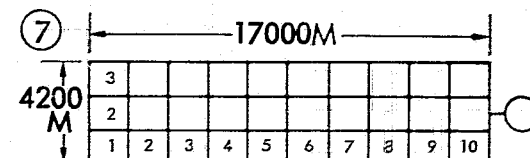
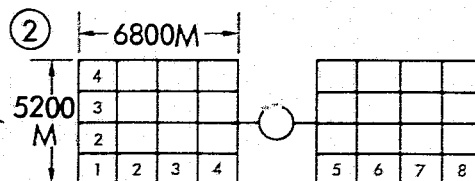
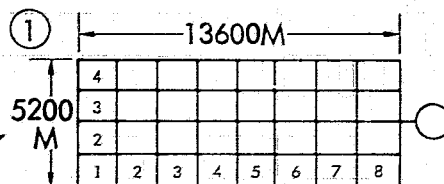
CONFIGURATION OPTIONS

This chart identifies eight (8) different configuration options compared in the trade study. The major comparisons showed the impact from: solar cell material selection (GaAs and Silicon), concentration ratio, antenna mounting position (end and center), radiation annealable and non-annealable assumptions, and solar array width and length ratio.

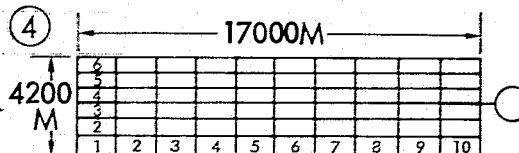
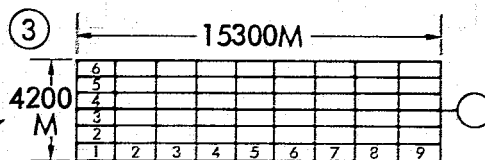
For the study, solar cell and power distribution efficiencies were held constant at the values shown in the lower right.

CONFIGURATION OPTIONS

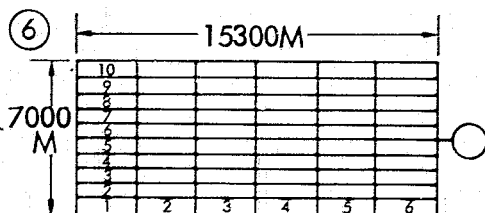
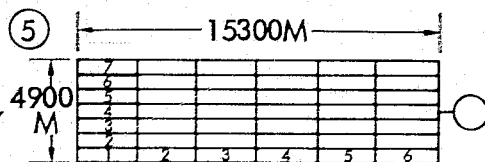
REFERENCE CO-PLANAR
(GaAs CR=2 SELF ANNEALING)
END MT. ANTENNA
CENTER MT. ANTENNA



GaAs CO-PLANAR
(CR=1, END MOUNTED ANT.)
SELF ANNEALABLE
NON ANNEALABLE



SILICON CO-PLANAR
(CR=1, END MT. ANTENNA)
INDUCED ANNEALABLE
NON ANNEALABLE



NOTE:

CELL EFFICIENCY

GaAs = 20% (AMO, 28C)

Si = 17.3% "

POWER DISTR. EFFICIENCY

$\eta_{LL} = 94\%$



CELL EFFICIENCY

Various parameters are compared to show relative comparisons between GaAs and Silicon solar cells. One important comparison is the cell efficiency. This chart reviews present technology status and shows a projected efficiency for GaAs of 22.1% based on best data obtained to date. This efficiency is compared to silicon at 14.5%. The SPS solar array design is based on GaAs efficiency of 20% and silicon efficiency of 17.3%. To achieve these operating efficiencies major technology advancements are required, particularly, considering the requirement of low solar cell cost.

CELL EFFICIENCY

GaAs

BEST DATA OBTAINED
TO DATE

$$I_{SC} = 34 \text{ MA/CM}^2$$

$$V_{OC} = 1.0 \text{ VOLTS}$$

$$\text{F.F.} = 0.88$$

$$\text{CALC } \eta = 22.1\%$$

$$\eta = \frac{I_{SC} V_{OC} \text{ F.F.}}{S}$$

REFERENCE: ROCKWELL SCIENCE CENTER

- RELATIVELY INSENSITIVE TO CELL THICKNESS
- SPS USES 20% AMO (28°C)
~1 MIL CELL WITH
5 μM ACTIVE CELL MAT'L THICK

SILICON

HIGHEST OUTPUT FOR THIN (1 MIL)
SILICON CELL = 78.5 MW/4 CM²
MEASURED $\eta = 14.5\%$

INTEGRATES OPTIMIZATION TECHNIQUES:

- ✓ • SHALLOW FUNCTIONS WITH OPT GRID PATTERNS
- ✓ • BACK SURFACE FIELDS
- ✓ • TEXTURED FRONT SURFACES
- ✓ • BACK SURFACE REFLECTORS
- ✓ • HIGHER RESISTIVITY (50 OHM-CM)

REFERENCE: SPECTROLAB

- MAXIMUM OUTPUT ACHIEVED AT 5 MILS*
- SPS USES 17.3% AMO (28°C) ~15.8% + 10%
IMPROVEMENT
~2 MIL CELL MAT'L THICK

*IF BACK SURFACE FIELD AND SURFACE REFLECTOR
TECHNOLOGY IS EMPLOYED - SPECTROLAB

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EFFICIENCY CHAIN SOLAR CELLS

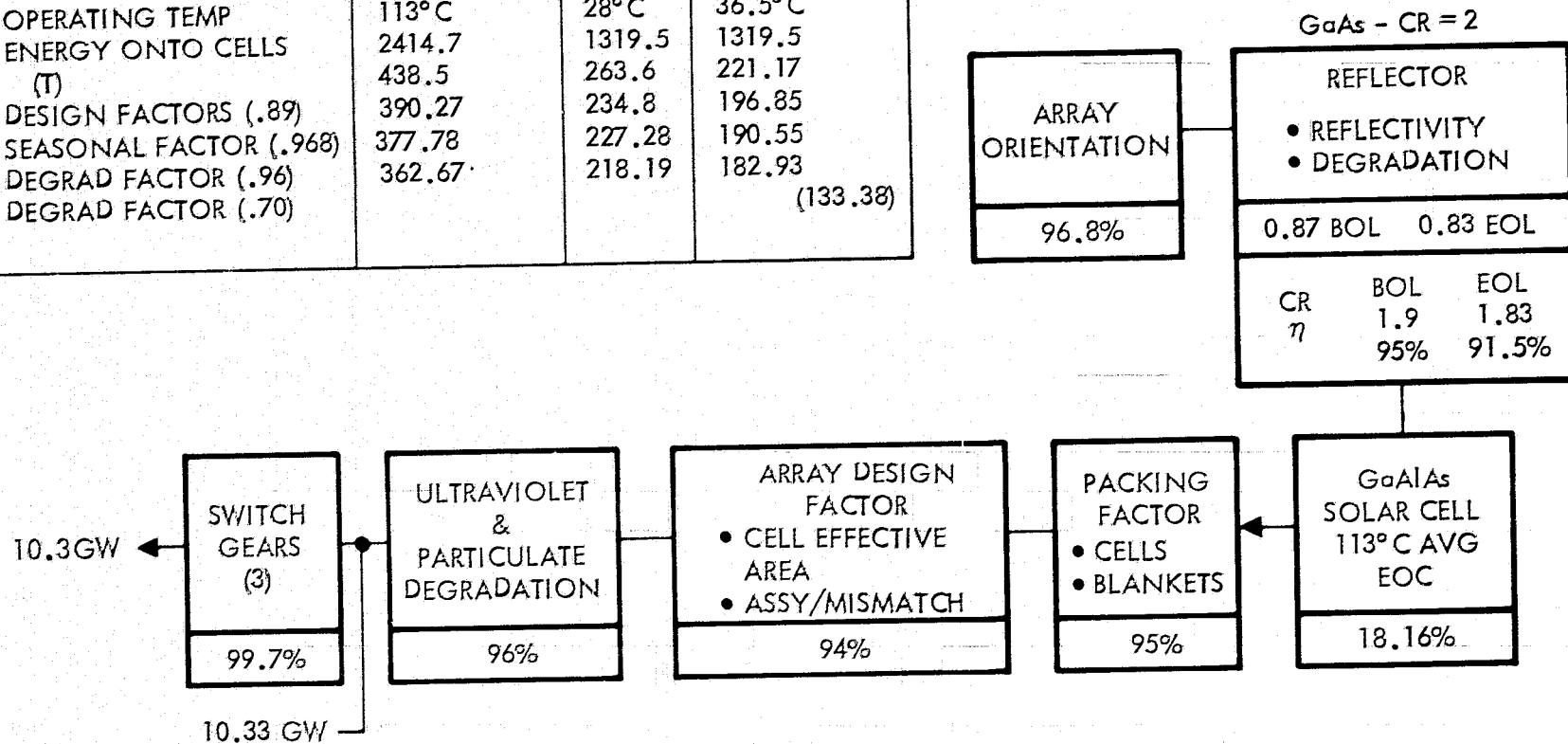
An updated SPS solar array efficiency chain is shown. These values have been corrected for improved solar cell performance as a result of the satellite being tipped 9.02° during summer solstice, thereby, reducing the inclination angle to the sun from 23.5° to 14.48° . The comparative solar array specific power outputs for GaAs (CR=1, CR=2) and silicon (CR=1) are shown in the table. These values were used to size the solar array in the trade off configuration comparisons.

A continuous review of subsystem efficiencies is maintained in order to provide updated efficiency factors for the design of the SPS. The summer solstice is taken as the sizing requirement since power output is a minimum during this period.

EFFICIENCY CHAIN SOLAR CELLS

SUMMER SOLSTICE (WATTS/M ²)			
	GaAs (20%)		SILICON (17.33%)
	CR=2	CR=1	CR=1
SOLAR INPUT	1319.5 W/M ²		→
OPERATING TEMP	113°C	28°C	36.5°C
ENERGY ONTO CELLS	2414.7	1319.5	1319.5
(T)	438.5	263.6	221.17
DESIGN FACTORS (.89)	390.27	234.8	196.85
SEASONAL FACTOR (.968)	377.78	227.28	190.55
DEGRAD FACTOR (.96)	362.67	218.19	182.93
DEGRAD FACTOR (.70)			(133.38)

• SOLAR CELL EFFICIENCY CHAIN



EFFICIENCY CHAIN - ROTATING ANTENNA AND GROUND RECEPTION

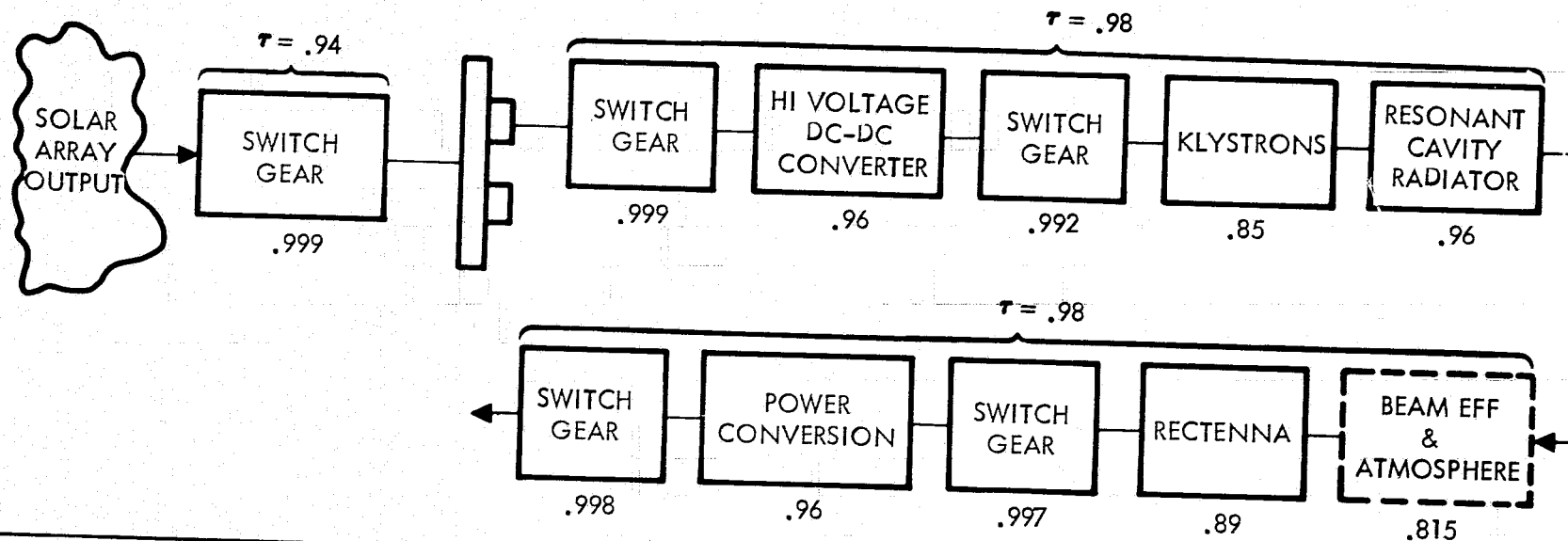
This chart shows the device/element power efficiency estimates established to date. The table shows klystron power output in kilowatts/klystron matched against thermal and power densities. The current Rockwell antenna layout was based on 52 kW/klystron and at the system efficiencies estimated (e.g., Exh. A/B 3rd Quarter), the system was sized to deliver 5.0 GW at the utility interface. Introducing updated estimates of system efficiencies results in less power output at the utility interface. The design has been modified and klystron power output increased to 56.5 kW/klystron. This new value results in slightly higher power densities at both the transmitter and the ionosphere as shown in the table. A redistribution of klystrons at the center of the antenna would be required to reduce these levels.

EFFICIENCY CHAIN - ROTATING ANTENNA AND GROUND RECEPTION

SOLAR ARRAY OUTPUT	KLYSTRON (135,864)	RESONANT CAVITY RADIATOR	(1) UTILITY INTERFACE	(2) POWER DENSITY AT TRANSMITTER	POWER DENSITY AT IONOSPHERE	COMMENT
9.51 GW	52 KW/(7.07 GW) KLYSTRON	6.79 GW	4.61 GW	21 KW/M ²	23 MW/CM ²	EXH A/B (3RD QUARTER)
9.92	54.3 (7.39 GW)	7.09	4.81	21.9 KW/M ²	24 MW/CM ²	EXH A/B (4TH Q)
10.3	56.5 (7.676 GW)	7.36	5.0	22.9 KW/M ²	25 MW/CM ²	EXH C (1ST Q)

NOTE:

(1) CORRECTED TO NEW EFFICIENCY CHAIN

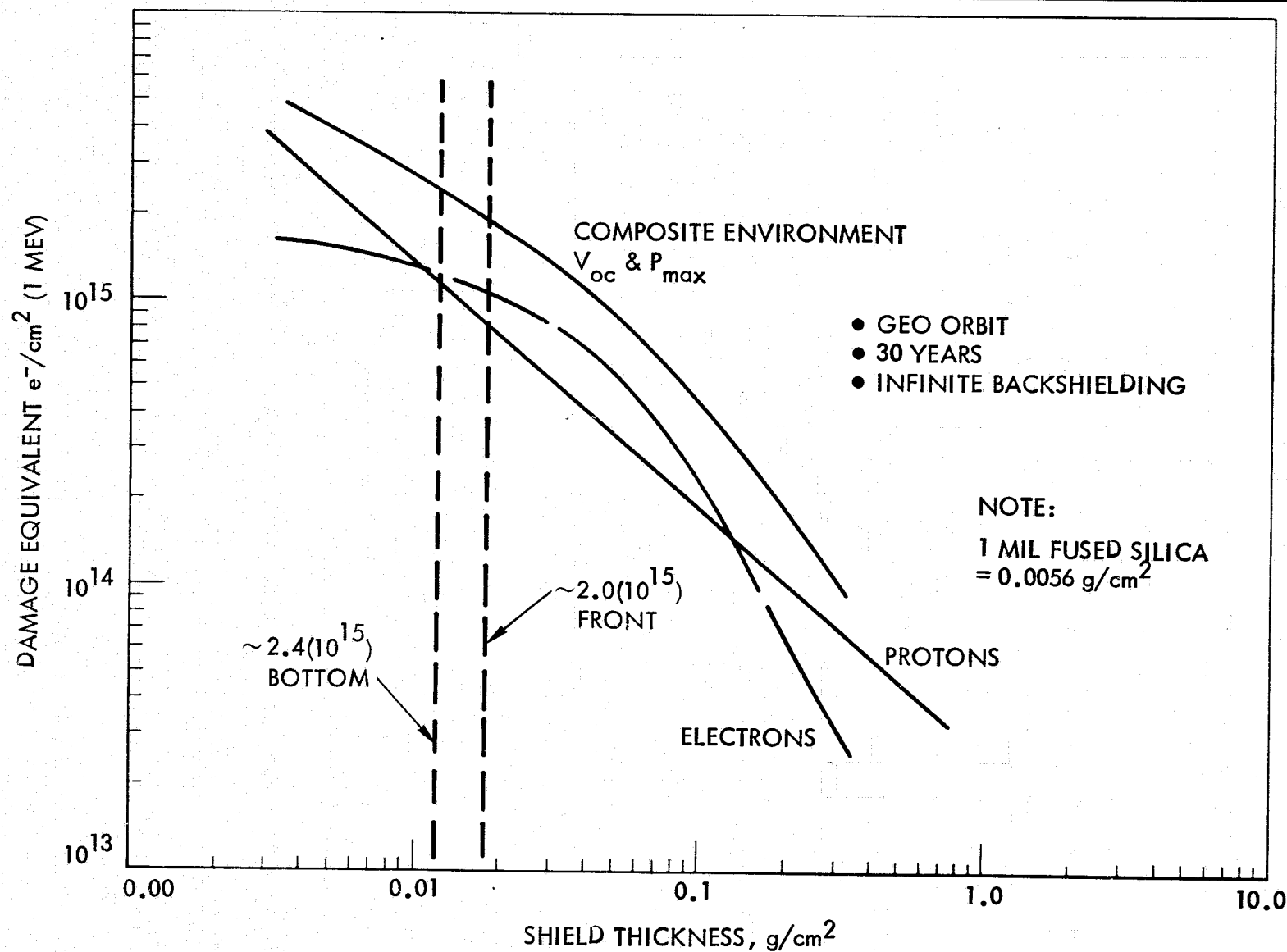


SOLAR CELL DAMAGE EQUIVALENT/MEV
ELECTRON FLUENCE VS SHIELD DENSITY

The solar array electrical output is affected by the on-orbit environment which includes trapped particle radiation, solar flare proton radiation, ultraviolet radiation, and the temperature cycling associated with the eclipse seasons. The natural trapped particle radiation environment was obtained from the "Solar Cell Radiation Handbook," TRW Report 21945-6001-RV-00. The trapped electrons are based on the AE-4 model of the outer radiation zone electron environment. For solar cells with minimal shielding, trapped protons may be neglected at geosynchronous orbit. The solar flare proton model was obtained by averaging the integral flux values for the five worst years of the 19th and 20th solar cycles. The values for damage equivalent 1 MEV electrons are taken from "A Proposal for Global Positioning Satellite Electrical Power Subsystem," General Electric, Space Division Proposal No. N-30065, 28 Feb. 1974.

The values of solar flare protons equivalent 1 MEV e- flux shown in the above report are multiplied by 1.35 to allow for the six quiet years of the solar cycle. The values for a 11 year cycle are then multiplied by a factor of 3 to obtain a 30 year model.

SOLAR CELL DAMAGE EQUIVALENT 1 MEV ELECTRON FLUENCE VS SHIELD DENSITY



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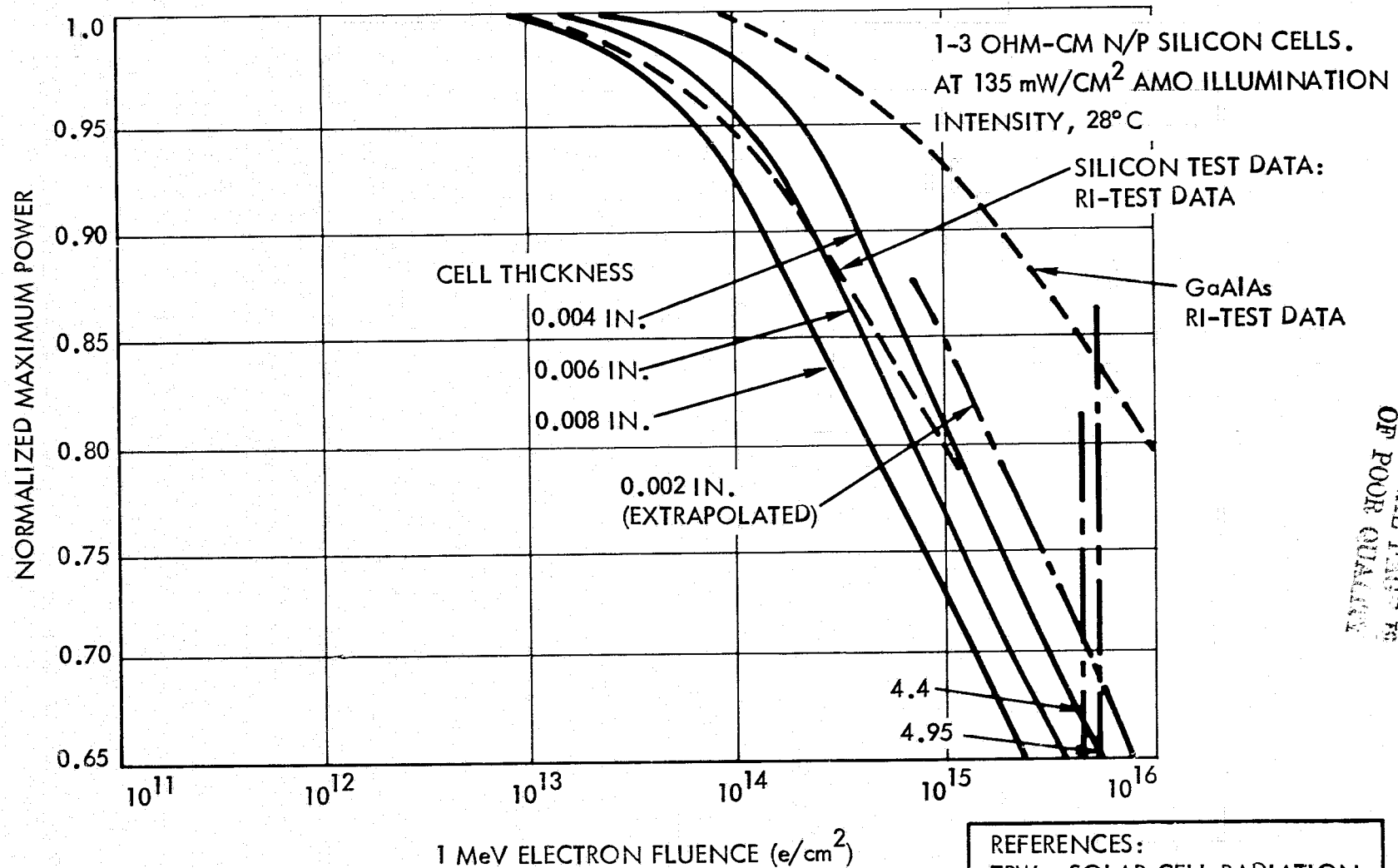
NORMALIZED MAXIMUM POWER VS. 1-MeV ELECTRON FLUENCE

This chart shows the effect of 1-MeV electron radiation on solar cell power for 1-3 ohm-cm N/P silicon solar cells of varying thickness. These data are from the "Solar Cell Radiation Handbook," TRW Report 21945-6001-RV-00. The data shown are a composite of data from several sources and represents the mean behavior of n-p conventional solar cell production in the United States. Solar cells produced with significant changes in composition may not show the same radiation loss factors.

The significance of the data shown is that thinner solar cells, percentage wise, show less radiation degradation than thicker cells. The data shown is obtained by dividing solar cell maximum power at each radiation fluence by the cell output (different for each cell thickness) before irradiation.

GaAlAs and silicon test data results from tests conducted by Rockwell are shown for comparison.

NORMALIZED MAXIMUM POWER VS 1-MEV ELECTRON FLUENCE



REFERENCES:
TRW - SOLAR CELL RADIATION
HANDBOOK
RI - TEST DATA

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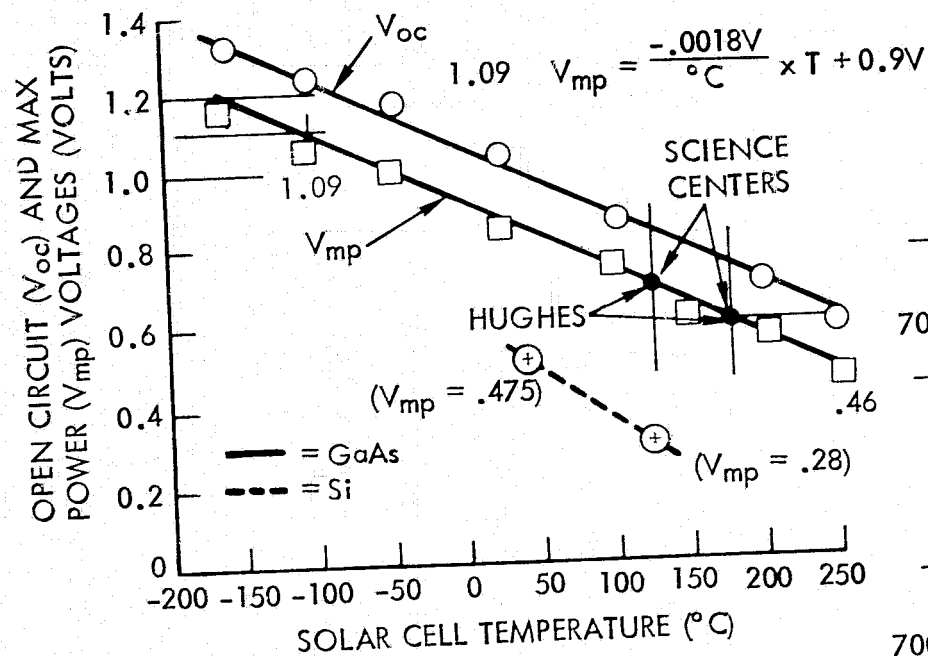


SOLAR PANEL BAY DIMENSIONS ~ VOLTAGE CONSIDERATIONS

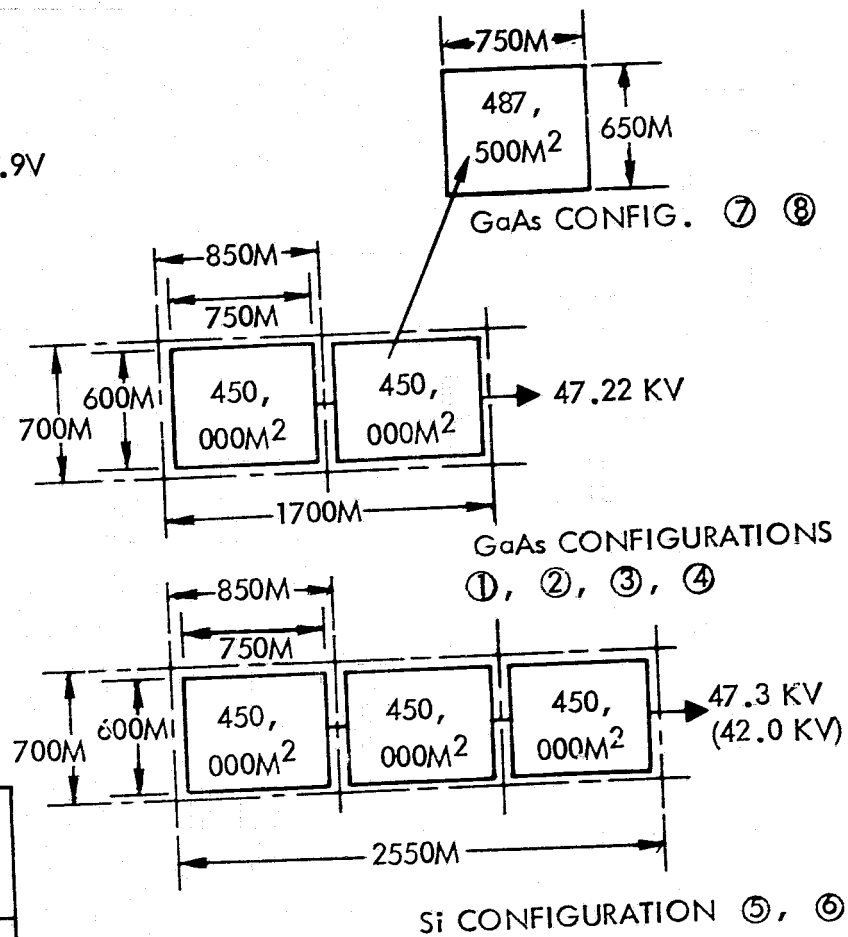
The open circuit voltage and maximum power voltage curves of the GaAlAs cells are shown in the figure. The curves are the experimental results of the Rockwell Science Center and the Hughes Research Center GaAlAs solar cell development programs. The relationship for the maximum power voltage is

$V_{mp} = -0.0018 \frac{V}{^{\circ}C} \times T + 0.9 \text{ V.}$ At the solar array operating temperature of $125^{\circ}C$, the maximum power voltage of the cell is 0.69 V. A reference V_{mp} for silicon is shown. These values for V_{mp} are used in determining overall blanket dimensions. A basic 750 meter length was selected to provide ~47.3 kV output. The width of the blanket is dictated by structural considerations and modular blanket widths of 25 meter rolls.

SOLAR PANEL BAY DIMENSIONS ~ VOLTAGE CONSIDERATION



CELL TYPE	OPS TEMP.	INITIAL CELL VOLTAGE	E.O.L. CELL VOLTAGE	VOLTAGE PER METER
GaAs	113 $^{\circ}C$	.69	.656	31.48
Si	36.5 $^{\circ}C$	.475	.438	21.02



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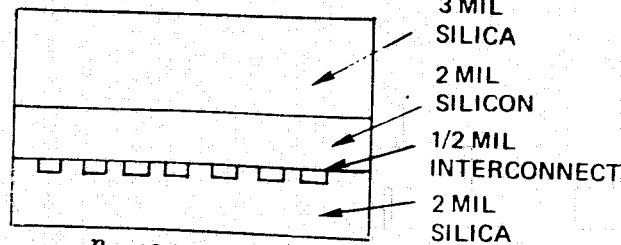
SOLAR CELL CONFIGURATIONS

A comparison of a silicon solar cell with GaAlAs solar cells is presented in the chart. The Si cell for advance applications are projected to weigh from 0.53 to 0.338 kg/m². For reference purposes, present state-of-the-art silicon cells weigh 0.936 kg/m². Weight improvements for Si cells are from use of thinner cells (4 to 2 mils thick) and thinner covers. Boeing used a weight of 0.427 kg/m² in their SPS study for NASA-LBJSC.

GaAlAs solar cells are used in the baseline for trade off studies. Weights are projected from 0.304 to 0.228 kg/m². In the study, the weight statements used a weight of 0.252 kg/m². The GaAlAs cells are presently under development and appear to offer a high potential for low weight, high efficiency and mass production capabilities. A particularly unique configuration is the inverted GaAs/Sapphire where the transparent Sapphire is utilized as the cover; thereby, removing a basic weight item and allowing the cell to operate at elevated temperatures. The point design solar array SPS utilizes the inverted GaAlAs/Sapphire cell concept.

SOLAR CELL CONFIGURATIONS

SILICON

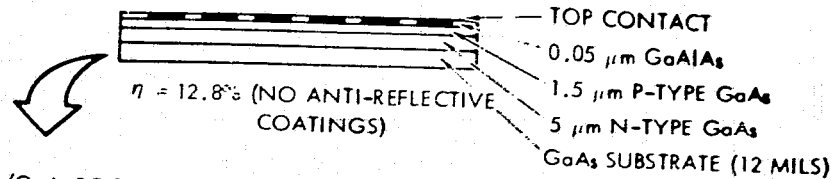


$\eta = 18.0$ AT 25C
0.427 Kg/M²

3 MIL SILICA	16.76 MG/CM ²
2 MIL SILICON	11.58
1/2 MIL INTER	1.14
2 MIL SILICA	11.18
	40.66
5%	2.03
	42.69
	(0.427 Kg/M ²)*

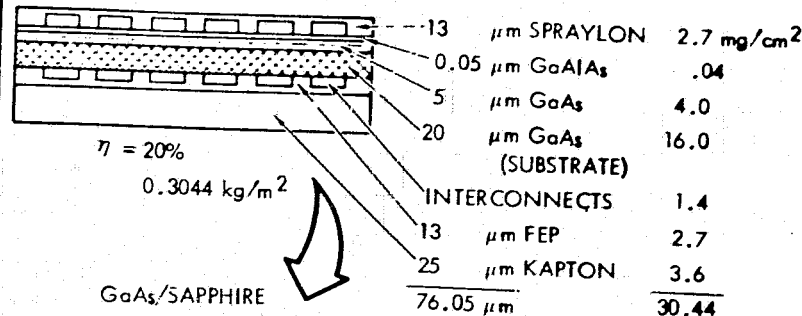
*BOEING REF

ERD - EXPERIMENTAL GaAlAs CELL



$\eta = 12.8\%$ (NO ANTI-REFLECTIVE COATINGS)

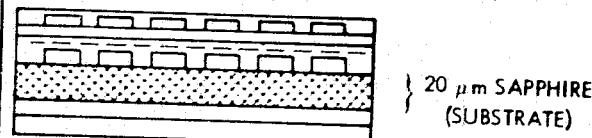
GaAs/GaAs ROCKWELL SPS



$\eta = 20\%$

0.3044 kg/m²

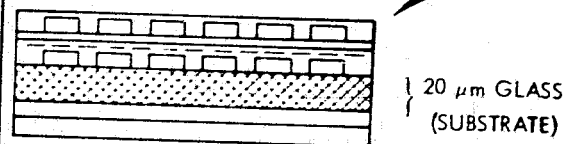
GaAs/SAPPHIRE



$\eta = 20\%$

0.2794 kg/m²

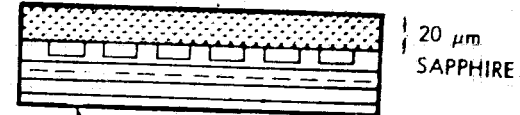
GaAs/GLASS



$\eta = 10$ TO 14%

0.2282 kg/m²

INVERTED GaAs/SAPPHIRE



$\eta = 19-20\%$

0.252 kg/m² *

POINT DESIGN CONFIGURATION

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CONFIGURATION OPTION COMPARISON DATA

This chart presents the summary comparison data on the eight configuration options studied. Option 1 and 2 is GaAs CR=2 for end mounted and center mounted antennas. Option 6 and 7 differ from 1 and 2 in that a narrow width is utilized (i.e., 3 trough wide vs 4). Options 3 and 4 are the GaAs CR=1 for annealable and nonannealable considerations. Options 4 and 5 are silicon CR=1 for annealable and nonannealable. The First Quarter baseline (i.e., 2 tier configuration) is presented for reference only.

Conclusions reached from the data is that the difference between 4 trough width and 3 trough width in terms of weight is negligible ($<400,000$ kg). However, there is a significant weight savings for a center mounted antenna ($\sim 2.0 \times 10^6$ kg). This is due to the savings in power distribution weight. A small difference in weight is shown between annealable and nonannealable GaAs CR=1. Major weight differences are shown for the silicon vs gallium arsneide and for annealable vs nonannealable silicon.

CONFIGURATION OPTION COMPARISON DATA

CONFIGURATION	①	②	③	④	⑤	⑥	⑦	⑧	FIRST Q BASELINE
CELL MATERIAL	GaAs				Si		GaAs		
CONC RATIO	2		1				2		
ANT MOUNT	END	CENTER	END					CENTER	
RADIATION DEGRAD FACTOR	.96			.84	.96	.70	.96		
CELL OUTPUT (W/M ²)	362.7		218.2	190.9	182.9	133.4	362.7		336.6
SOLAR CELL AREA (10 ⁶ M ²)	28.4		47.2	53.95	56.3	77.2	28.4		30.6
REFLECTOR AREA (10 ⁶ M ²)	57.6						57.6		61.2
BAY DIMENSIONS (M)	700 X 1700				700 X 2550		750 X 1700		(650)
NUMBER SOLAR CELL BAYS	4 X 8 = 32		6 X 9 = 54	6 X 10 = 60	7 X 6 = 42	10 X 6 = 60	3 X 10 = 30		600 X 1600
REFL BAY DIMENSION (M)	1300 X 1700						1400 X 1700		3 X 12 = 36
PLAN FORM (M)	5200 X 13600		4200 X 15300	4200 X 17000	4900 X 15300	7000 X 15300	4200 X 17000		(1350)
PLAN FORM AREA (10 ⁶ M ²)	70.72		64.3	71.4	74.97	107.1	71.4		1250 X 1600
NO. SWITCHGEAR (ON ARRAY)	1120		1890	2100	2436	3480	1140		3850 X 19200
COLLECTOR ARRAY (10 ⁶ KG)									73.92
STRUCTURE & MECH	2.156		2.043	2.168	2.230	2.791	2.156		1260
SOLAR PANELS	7.258		11.897	13.595	23.988	34.506	7.258		
SOLAR REFLECTORS	1.182						1.182		3.825
POWER DISTRIBUTION	2.719	1.55	2.879	2.545	3.058	3.87	3.03	1.272	7.722
ATT CONTROL/IMS/ROT JT	.385		.350	.389	.408	.583	.385		1.108
	(13.699)	(12.53)	(17.259)	(18.697)	(29.684)	(41.750)	(14.011)	(12.253)	1.081
ANTENNA SECTION	16.297								.385
SUBTOTAL	29.996	28.827	33.556	34.994	45.981	58.047	30.308	28.55	(14.127)
25% GROWTH	7.499	7.206	8.389	8.749	11.495	14.512	7.577	7.137	
TOTAL	37.495	36.033	41.945	43.743	57.476	72.559	37.885	35.687	30.424
									7.606

ORIGINAL PAGE 13
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SOLAR CELL SYSTEM COMPARISON SUMMARY

This chart shows a comparison summary for each of the parameters studied, i.e., efficiency, radiation degradation factor, annealing, cell temperature coefficient, weight, and cell area.

SOLAR CELL SYSTEM COMPARISON SUMMARY

PARAMETER	GaAs	SILICON
EFFICIENCY	✓ BEST DATA OBTAINED TO DATE CALCULATES ~ 22.1% AMO (28C) 20%	HIGHEST OUTPUT FOR THIN (1 MIL) MEASURES ~ 14.5% AMO (28C) 17.3%
RADIATION DEGRADATION FACTOR	✓ 0.84 — .96 NON- ANNEALABLE	0.70 — .97 NON- ANNEALABLE
ANNEALING	✓ SELF ANNEALING AT ≥ 125 C	INDUCED ANNEALING AT ≥ 500 C
CELL TEMP. COEFFICIENT	✓ $\frac{-.00175 \text{ V}}{^{\circ}\text{C}}$; VMP = .65 (125C)	$\frac{-.00215 \text{ V}}{^{\circ}\text{C}}$; VMP = .28 (125C)
WEIGHTS	✓ 0.252 KG/M ² (BLANKET) CR = 1 END MT ANTENNA 41.945 (10 ⁶) KG ANNEALABLE 43.743 (10 ⁶) KG NONANNEALABLE CR = 2 END MT ANTENNA 37.495 (10 ⁶) KG 4 TROUGH 37.885 (10 ⁶) KG 3 TROUGH CR = 2 CENTER MT ANTENNA 36.033 (10 ⁶) KG 4 TROUGH 35.687 (10 ⁶) KG 3 TROUGH	0.427 KG/M ² (BLANKET) 57.476 (10 ⁶) KG ANNEALABLE 72.559 (10 ⁶) KG NONANNEALABLE - - - -
CELL AREA	✓ 28.4 X 10 ⁶ M ² (10.3 GW) 362.7 W/M ² (CR = 1.83)	56.3 X 10 ⁶ M ² (10.3 GW) 182.9 W/M ² (CR = 1)

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PRELIMINARY TRADE RESULTS

The preliminary trade results show that GaAs solar cells remain the preferred cell material (compared to silicon). This is based on its higher efficiency (20% vs 17.3%); lower space radiation degradation (16% vs 30%); greater potential for self-annealing out of radiation damage (125°C threshold temperature vs >500°C); lower specific weight (0.252 kg/m² vs 0.427 kg/m²); its compatibility with concentrators; improved temperature coefficient; smaller cell area, potential for cell efficiency improvement (the multiple band gap concept is essentially a gallium arsenide cell with potential of 25-30% cell efficiency) and lower overall SPS cost.

Silicon offers advantages in that silicon material is easily available; however, scale up is required to obtain the necessary quantity of semi-grade silicon (e.g., 11850 metric tons per 5 GW). A more mature silicon technology exists. Gallium arsenide solar cell development will require a new industry in reclaiming gallium from bauxite and in producing the required solar cells ~30×10⁶m² per 5 GW.

The center mounted configuration is the lowest weight of those studied. This is due to the savings in the power distribution resulting from shorter conductor lengths.

The remainder of results are self-explanatory.

PRELIMINARY TRADE RESULTS

- GaAs SOLAR CELL REMAINS "BEST" CELL MATERIAL (COMPARED TO SILICON)
 - HIGHER CONVERSION EFFICIENCY (20% VS 17.3%)
 - LESS SPACE RADIATION DEGRADATION (.84 VS .70)
 - GREATER POTENTIAL FOR SELF ANNEALING OF RADIATION DAMAGE (125°C VS 500°C)
 - LOWER SPECIFIC WEIGHT (0.252 KG/M² VS 0.427 KG/M²) (1)
 - COMPATIBLE WITH CONCENTRATION (CR=2 VS CR=1)
 - IMPROVED TEMPERATURE COEFFICIENT (-0.0017V/°C VS -0.00215V/°C)
 - SMALLER SOLAR CELL AREA (28.4 × 10⁰ M² VS 56.3 × 10⁶ M²)
 - POTENTIAL FOR CELL EFFICIENCY IMPROVEMENT (MULTIPLE BANDGAP 25-30%)
 - LOWER SPS OVERALL COST (3130.2M VS 5257M NORMALIZED) (1)
- SILICON OFFERS SOME ADVANTAGES
 - BETTER MAT'L AVAILABILITY ASSURANCE (67000 MT OF SILICON REQUIRED TO PRODUCE 11850 MT SEMI CONDUCTOR GRADE SILICON (SeG-Si)
NOTE: APPROX. 9,750,000 M.T BAUXITE REQUIRED TO PRODUCE 390 MT GALLIUM FOR 1 SPS
 - MORE MATURE SOLAR CELL INDUSTRY (GALLIUM ARSENIDE SOLAR CELLS REQUIRE NEW INDUSTRY)
- GaAs CR=2 CENTER MOUNTED ANTENNA OFFERS LOWEST WEIGHT SPS OF THOSE STUDIED. SIGNIFICANT POWER DISTRIBUTION WEIGHT SAVINGS RANGE ~ 1.5 × 10⁶ KG TO 1.8 × 10⁶ KG
- SILICON CR=1 SPS WEIGHT PENALTY RANGE 21.8 × 10⁶ KG TO 36.9 × 10⁶ KG
- GaAs CR=1 SPS WEIGHT PENALTY RANGE 6.26 × 10⁶ KG TO 8.06 × 10⁶ KG

NOTE: (1) TAKEN FROM 1ST QUARTERLY

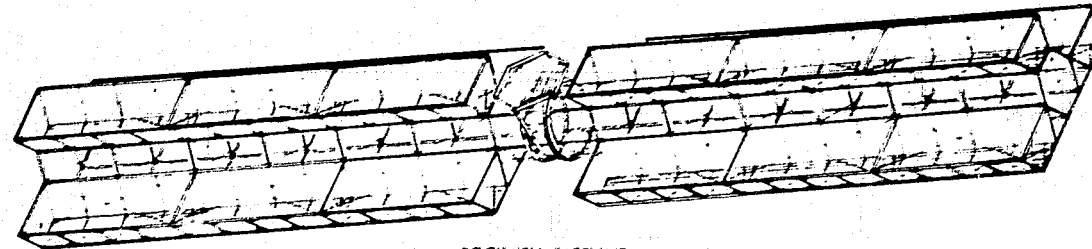


ANTENNA DESIGN CONCEPTS

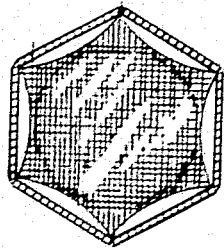
Thermal response considerations will be a significant factor in the selection of SPS antenna configuration and location. The chart depicts a number of the primary concepts evaluated for possible ultimate implementation. Both compression frame - tension web structures and space frame truss networks were investigated in combination with center mounted and end mounted antennas.

ANTENNA DESIGN CONCEPTS

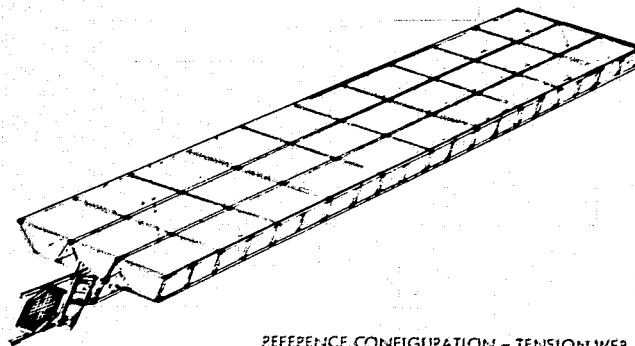
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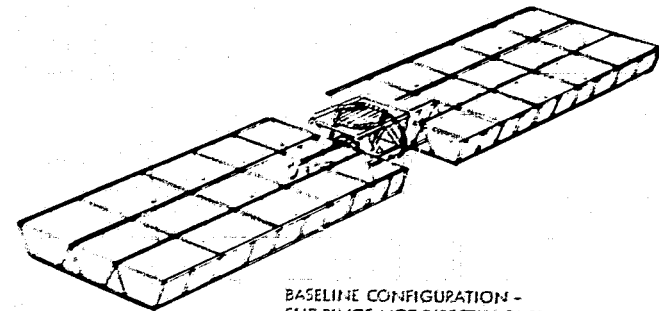
ROCKWELL BASELINE



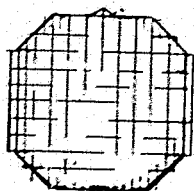
TENSION WEB



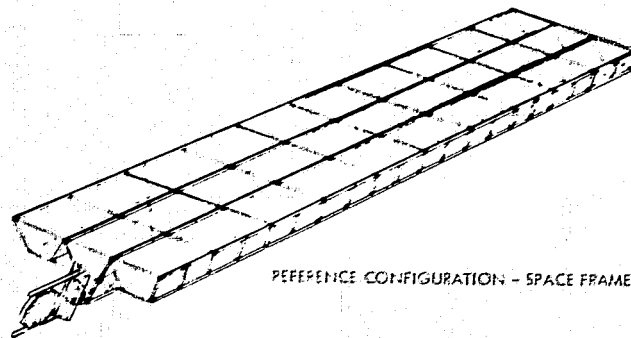
REFERENCE CONFIGURATION - TENSION WEB



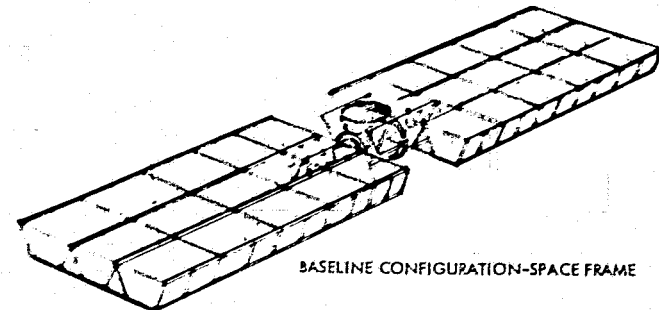
BASELINE CONFIGURATION -
SLIP RINGS NOT DIRECTLY OVER
ANTENNA



SPACEFRAME



REFERENCE CONFIGURATION - SPACE FRAME



BASELINE CONFIGURATION-SPACE FRAME



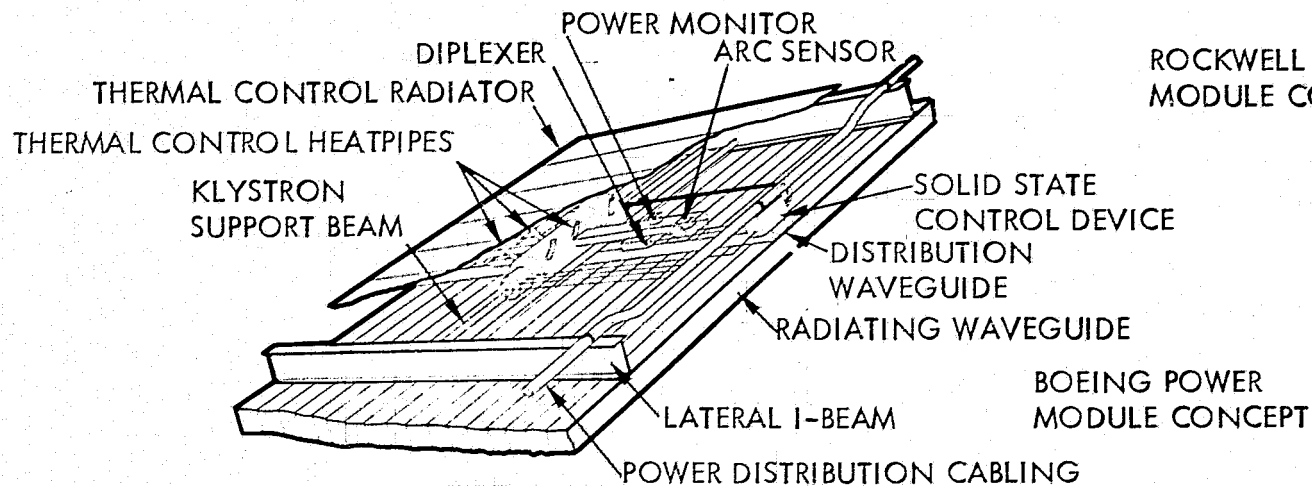
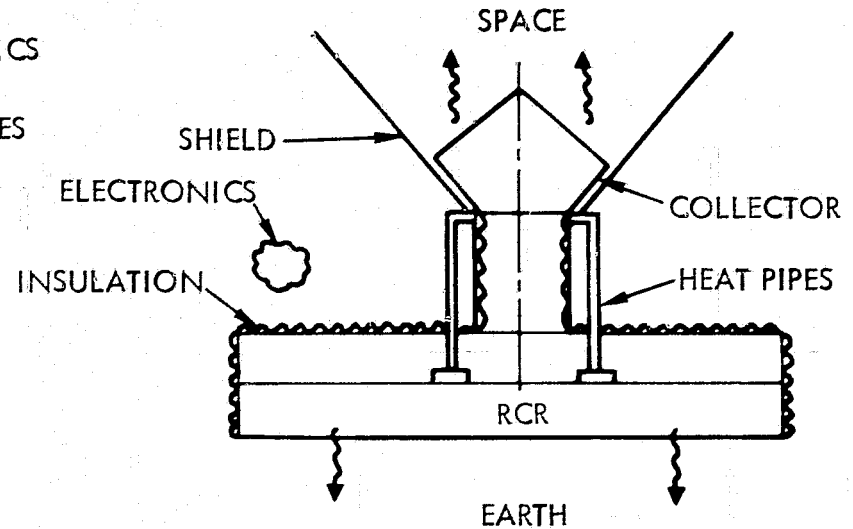
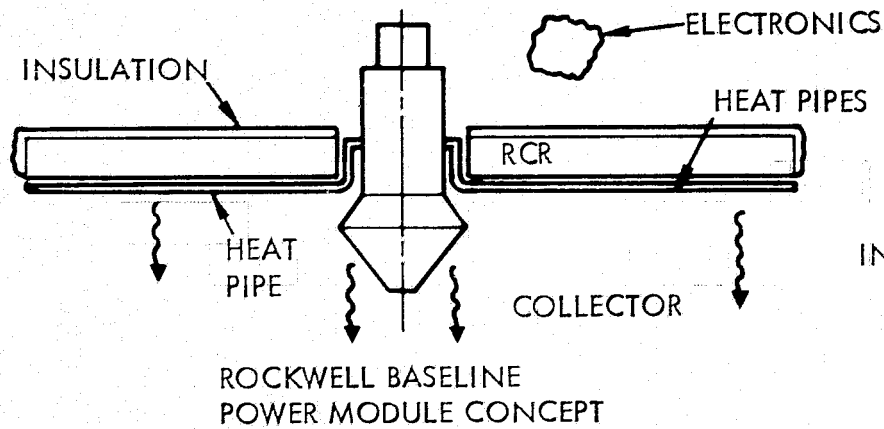
KLYSTRON DESIGN CONCEPTS

The primary candidate klystron study designs are illustrated. The initial Rockwell baseline power module was selected during Phase 1 because of thermal rejection considerations relative to potential overheating of the slip rings, temperature limitations on the electronics and the postulated requirement (later deleted) for manned access to the antenna during operation. The alternate Rockwell concept, which has been adopted during Phase 2, eliminates the performance uncertainty and efficiency losses resulting from poking the klystron tube through the radiator. This selection follows from analyses which demonstrate that the slip rings can be thermally protected and that the weight penalty associated with controlling electronic component temperatures is relatively small. This klystron concept appears more advantageous than the Boeing concept because of its less complex and lighter weight thermal control system. In addition, it appears more flexible in terms of distributing the heat to be rejected in both directions (space and earth).

KLYSTRON DESIGN CONCEPTS

DISTRIBUTED EARTH AND SPACE HEAT REJECTION CONCEPT

EARTH SIDE HEAT REJECTION CONCEPT



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GRAPHITE COMPOSITE MATERIALS TEMPERATURE LIMITS

Potential graphite fiber reinforced resin systems that might be utilized on large space structures are listed. The maximum use temperatures shown are based upon GD quoted limits and Rockwell estimates. Further testing, including extended period outgassing determinations, could result in modification of these values.

GRAPHITE COMPOSITE MATERIALS TEMPERATURE LIMITS

MATERIAL	MAX. ALLOWABLE TEMP. °K
THERMOSETTING RESINS	
EPOXY	395
PHENOLIC	435
POLYIMIDE	
ADDITION	475
CONDENSATION	560
THERMOPLASTIC MATERIALS	
POLYSULFONE	380
POLYIMIDE	590



SPACE FRAME CONFIGURATION LIMITATIONS OF LOW TEMP. COMPOSITES

In space frame configurations the primary structure is in close proximity to the high temperature, heat rejection surface of klystrons (assuming the tube is not poked through). Independent studies by Rockwell, General Dynamics, and Grumman have demonstrated that temperatures in the central region of the antenna will exceed allowables for low temperature composites. For beam machine concepts this can complicate the construction scenario by potentially requiring the substitution of a different machine at some point in the assembly sequence. This problem is illustrated in the chart for the candidate polysulfone resin assuming a "worst case" condition. In this environment the temperature could exceed the allowable maximum even if the klystron heating is neglected.

For compression frame-tension web structures this situation can not occur. The frame will experience relatively low temperatures and the secondary structure is sufficiently removed from the center of the antenna to permit application of low temperature composites. This represents a substantial thermal advantage for tension web structures.

SPACE FRAME CONFIGURATION LIMITATIONS OF LOW TEMP. COMPOSITES

- ASSUME MAX. ALLOWABLE TEMP. = 380°K ; $\alpha_s = .9$ $\epsilon = .8$

FOR THESE PROPERTIES THE MAXIMUM ALLOWABLE EFFECTIVE ANTENNA TEMPERATURE IS (APPROXIMATELY):

$$(.9)(1352) + .8\sigma(T_{\text{MAX}}^4 - 400^4) = .8\sigma 400^4 \text{ OR } T_{\text{MAX}} = 395^{\circ}\text{K}$$

- FOR "WORST CASE" SOLAR LOADING ON ANTENNA (EVEN IF KLYSTRON EMITS ZERO POWER) ASSUMING RADIATOR HAS BLACK COATING AND ZERO CAPACITANCE STRUCTURE.

$$(.93)(1352) = .85\sigma T^4 \text{ OR } T_{\text{MAX}} = 402^{\circ}\text{K}$$

- ∴ UNDER WORST CASE ASSUMPTIONS LOW TEMP. COMPOSITES MAY BE MARGINAL FOR ZERO POWER KLYSTRONS.



50 KW KLYSTRON RADIATOR THERMAL LEVELS

Temperature extremes across the antenna (center to edge) are presented for the Phase 1 baseline and improved klystron concepts. The edge values assume that the total allowable radiator area is used. Temperatures at the center were calculated for no rear surface insulation or collector shield. If the shield is used (as shown in the earlier chart) then the maximum value would be approximately 100°C. The high proportion of waste heat that can be rejected from the earth side ($\approx 30\%$) represents a significant advantage of this concept.

50 KW KLYSTRON RADIATOR THERMAL LEVELS

BASELINE KLYSTRON
(TUBE POKED THROUGH)
EARTH SIDE RADIATOR

ANTENNA
CENTER

152°C

ANTENNA
EDGE

-4°C

IMPROVED KLYSTRON
(COLLECTOR RADIATES
IN SPACE DIRECTION)

EARTH SIDE
SPACE SIDE

129°C
104°C

-14°C
-23°C



ANTENNA CONFIGURATION MATERIALS SUMMARY

The thermal advantage of tension web configurations is apparent from the tabular summary. Regardless of antenna location (center or edge) low temperature composites (or aluminum) can be used for the structure. As shown earlier this also simplifies the construction scenario inasmuch as a change in material for the antennas is not required and reduces SPS system costs by eliminating advanced high temperature material development requirements.

ANTENNA CONFIGURATION MATERIAL SUMMARY

CONCEPTS	TENSION WEB	SPACE FRAME
BASELINE KLYSTRON/ANTENNA	LOW TEMP. COMP. ALUMINUM	LOW TEMP. COMP. ALUMINUM
IMPROVED KLYSTRON/ BASELINE ANTENNA	LOW TEMP. COMP. ALUMINUM *	HIGH TEMP. COMP. HIGH TEMP. METALS
IMPROVED KLYSTRON/ BASELINE ANTENNA SLIP RINGS MOVED OUTWARD	LOW TEMP. COMP. ALUMINUM	HIGH TEMP. COMP. HIGH TEMP. METALS
BASELINE KLYSTRON/ REFERENCE ANTENNA	LOW TEMP. COMP. ALUMINUM	LOW TEMP. COMP. ALUMINUM
IMPROVED KLYSTRON/ REFERENCE ANTENNA	LOW TEMP. COMP. ALUMINUM	HIGH TEMP. COMP. HIGH TEMP. METALS

*HIGH TEMP. METALS FOR SLIP RINGS OR SLIP RINGS
PROTECTED BY THERMAL REFLECTORS



CONCLUSIONS

Results of the thermal study are summarized. Continued use of the compression frame - tension web structure and substitution of the improved, alternate Rockwell klystron concept for the previous baseline is recommended. Some additional evaluation of the relative merits of end mounted versus center mounted antennas appears warranted.

CONCLUSIONS

- MODIFIED (SLIP RINGS MOVED OUTWARD OR REFLECTOR PROTECTED) CENTER MOUNTED TENSION WEB ANTENNA IS BASICALLY THERMALLY EQUIVALENT TO END MOUNTED ANTENNA
- SPACE FRAME ANTENNA REQUIRES DEVELOPMENT OF HIGH TEMPERATURE COMPOSITE MATERIALS AND MORE COMPLEX CONSTRUCTION SCENARIO
- ALTERNATE ROCKWELL KLYSTRON CONCEPT IS LIGHTWEIGHT AND PERMITS SIMPLE THERMAL CONTROL DESIGN; ALSO ALLOWS VARYING PERCENTAGE OF WASTE HEAT REJECTED TOWARD EARTH FROM ABOUT 10% TO 30%.
- USE OF IMPROVED KLYSTRON CONCEPT WILL RESULT IN SOME WEIGHT PENALTY FOR ELECTRONICS



SPS STRUCTURAL ANALYSIS
THERMAL STUDY VERSUS MATERIAL SELECTION



OBJECTIVE

- PRESENT RESULTS OF THE THERMAL EFFECTS COMPARISON BETWEEN A METAL (ALUMINUM) AND AN ADVANCED COMPOSITE (GRAPHITE) MATERIAL DESIGN CONCEPT.
- MAKE KNOWN THE DEFICIENCIES OF NASTRAN COMPUTER PROGRAM FOR ANALYZING SPS STRUCTURES.

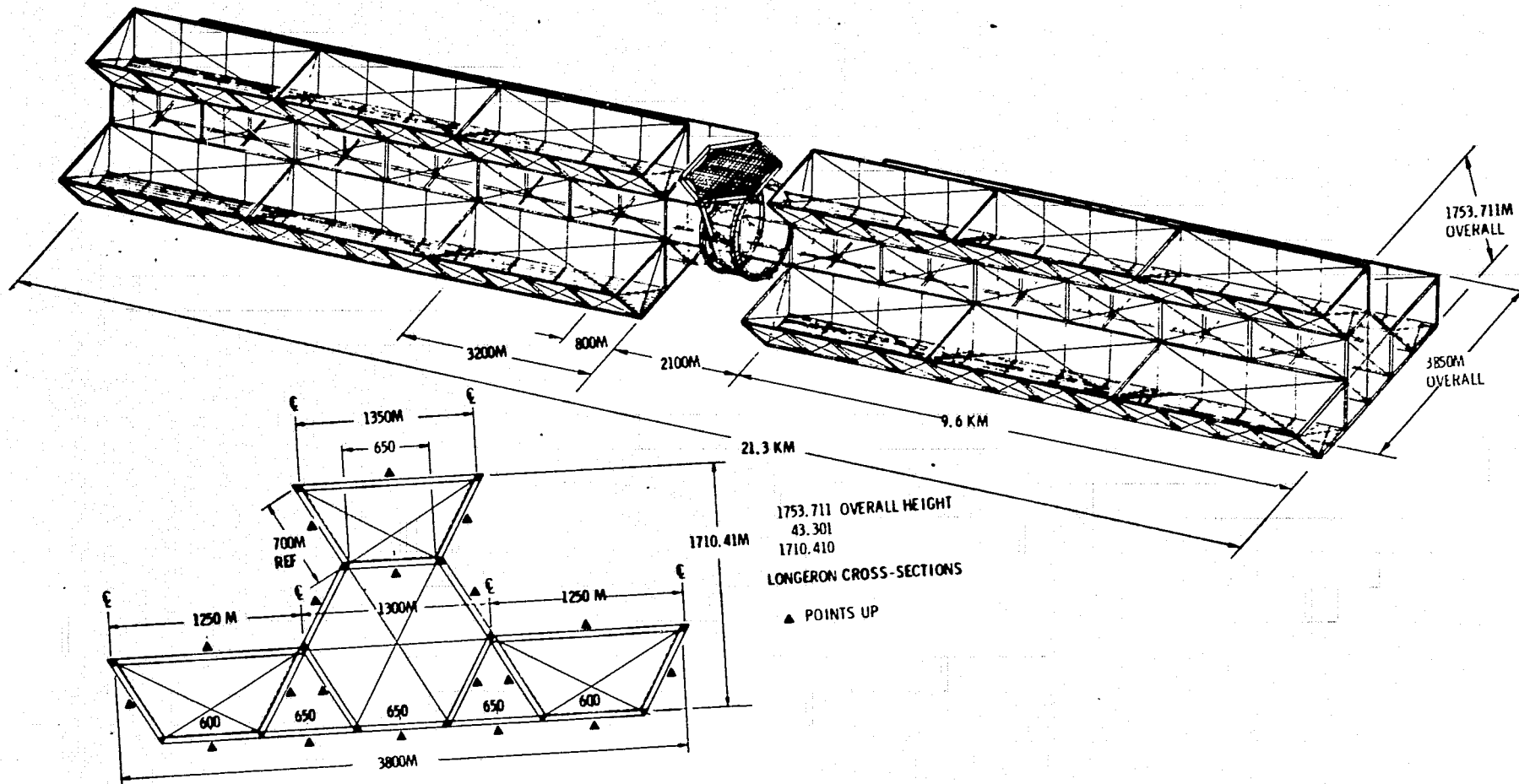
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SPS MODELING CONFIGURATION

This chart shows the Rockwell three-trough design configuration which was used for the thermal stresses and deflections analysis.

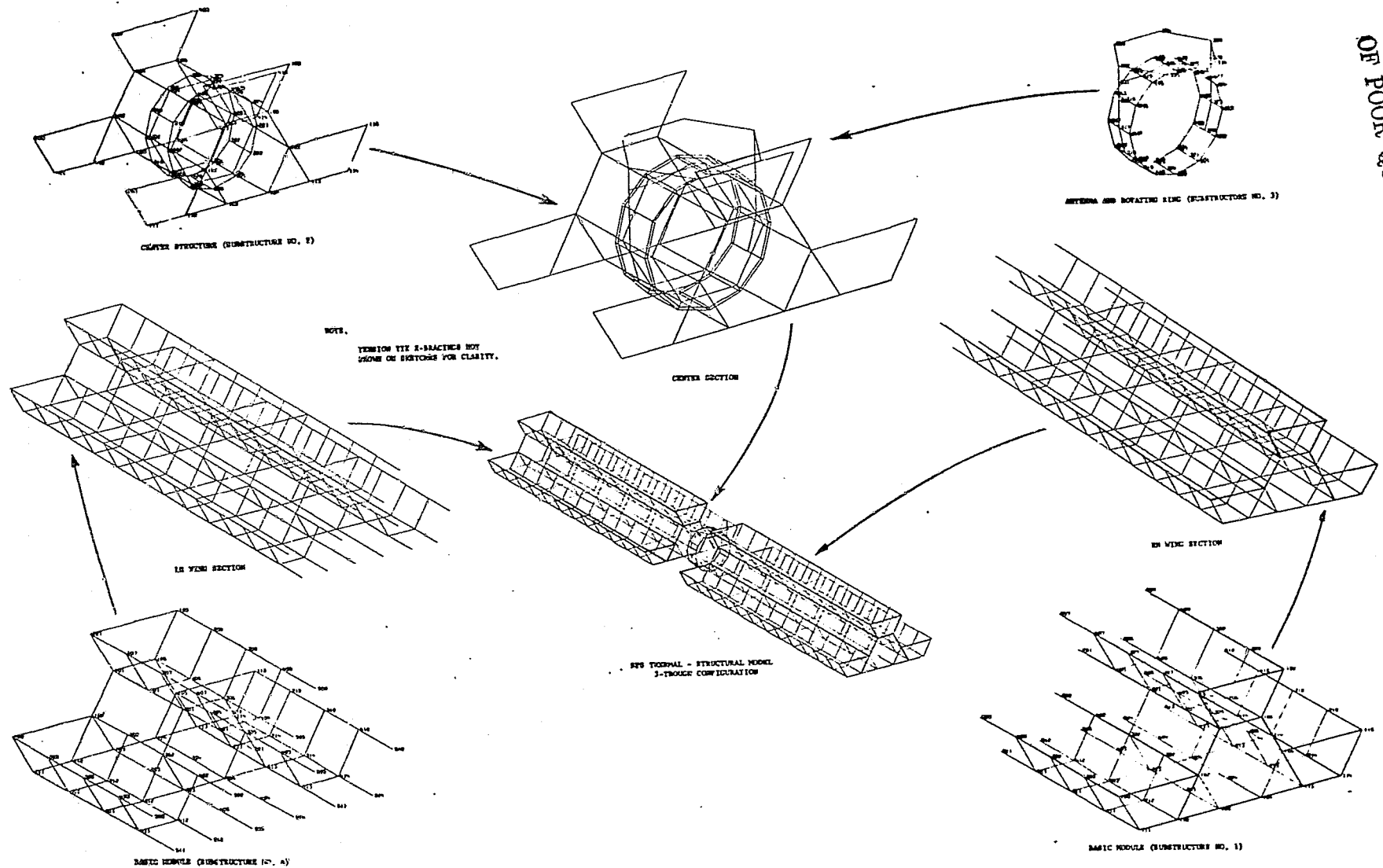
SPS MODELING CONFIGURATION



SPS COMPUTER PROGRAM STRUCTURAL MODEL

Using the substructure features of NASTRAN, the design configuration shown on the previous page was modeled. The sketch, using the CRT printcuts, depict the buildup of the SPS structure through the combination of the substructures. The right-hand wing is a combination of three translated Substructure No. 1 units. Similarly, the left-hand wing is a combination of three Substructure No. 4 units. The center structure is a combination of the center section, Substructure No. 2, and the antenna and rotating ring, Substructure No. 3. Finally, the center section, the left-hand wing, and right-hand wing are combined to form the SPS structural model.

SPS COMPUTER PROGRAM STRUCTURAL MODEL



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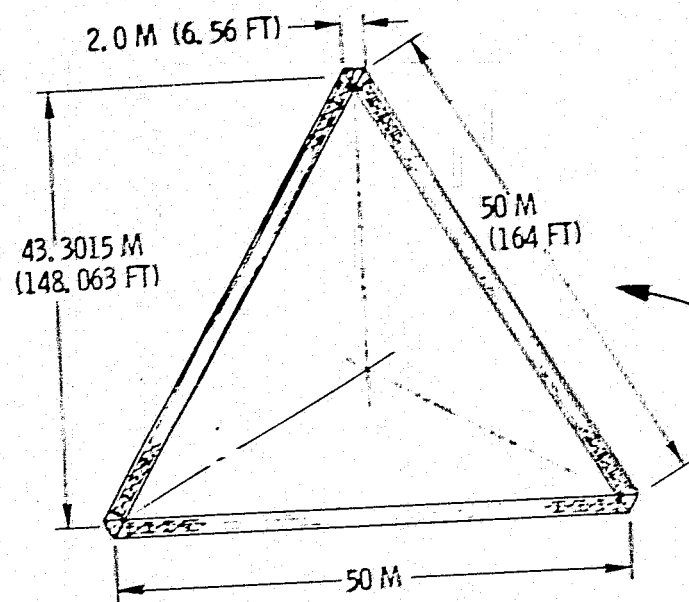


SPS STRUCTURAL CROSS-SECTIONS

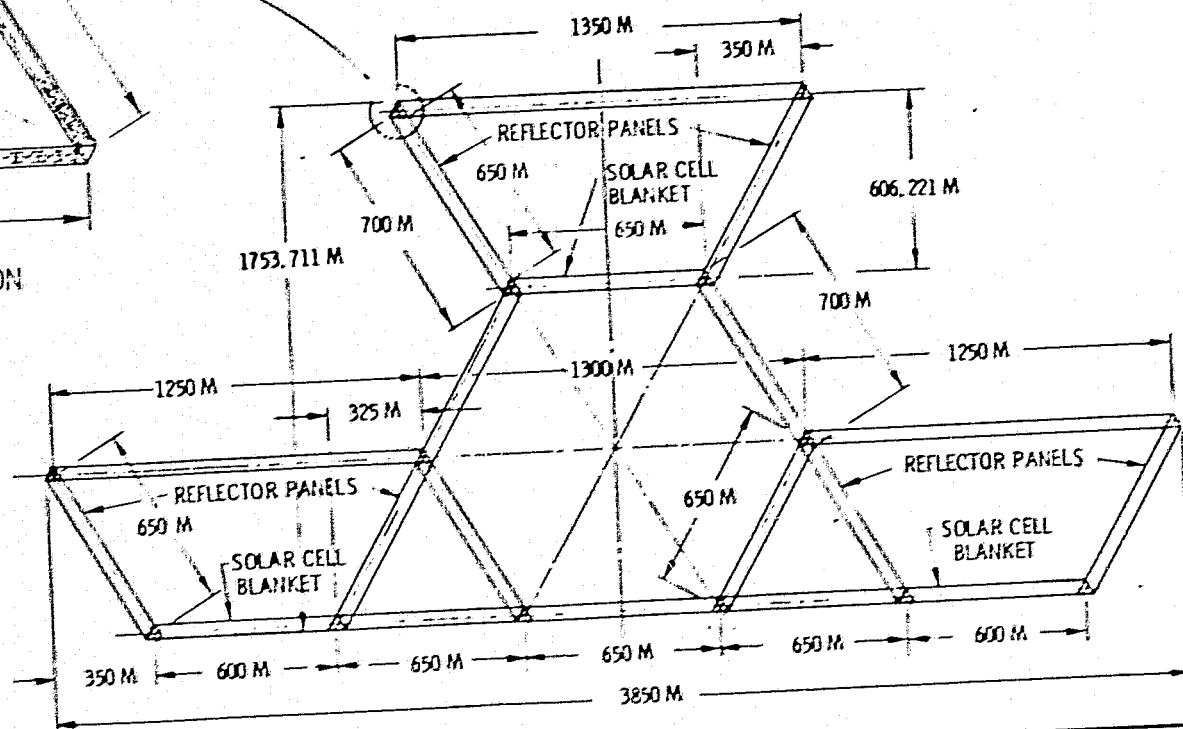
This chart shows an enlarged cross-section through the solar and reflector region, and a typical cross-section of the basic 50-m beam element used in the structural model. Although the 50-m beam is a structural subsystem of 2-m beams, the analysis did not consider the sublevel effects.

C-2

SPS STRUCTURAL CROSS-SECTIONS



50 M TRIBEAM GIRDER SECTION
TYPICAL ALL TRANSVERSE
& LONGITUDINAL BEAMS



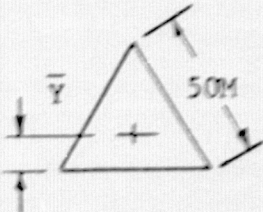
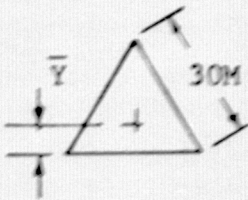
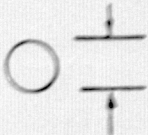
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SECTIONAL PROPERTIES

This chart summarizes the major sectional properties used in the computer program. Of particular notice is the replacement of the brush mechanism with equivalent spring member for connection between the rotating ring and inner support ring.

SECTIONAL PROPERTIES

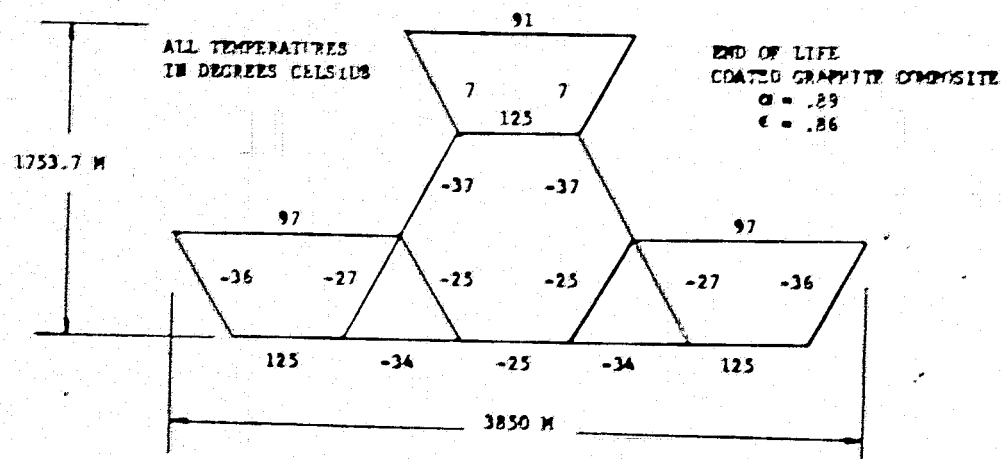
PROPERTY	TRIBEAM WING STRUCTURE 	TRIBEAM ANTENNA FRAME STRUCTURE 	TENSION TIE  $1.59 \times 10^{-3} \text{ M}$	EQUIVALENT BRUSH STRUCTURE (SPRING)
CROSS-SECTIONAL AREA	$369.3 \times 10^{-6} \text{ M}^2$	$0.81 \times 10^{-3} \text{ M}^2$	$1.98 \times 10^{-6} \text{ M}^2$	$2.32 \times 10^{-2} \text{ M}^2$
MOMENT OF INERTIA	0.154 M^4	0.126 M^4	-	1.54 M^4
POLAR MOMENT	$6.33 \times 10^{-2} \text{ M}^4$	$2.36 \times 10^{-2} \text{ M}^4$	-	0.60 M^4
CENTROID LOCATION ($\bar{Y}$)	14.43 M	8.82 M	-	-
AXIAL STIFFNESS ($K = EA/L$)	-	-	-	$5.97 \times 10^5 \text{ N/M}^2$



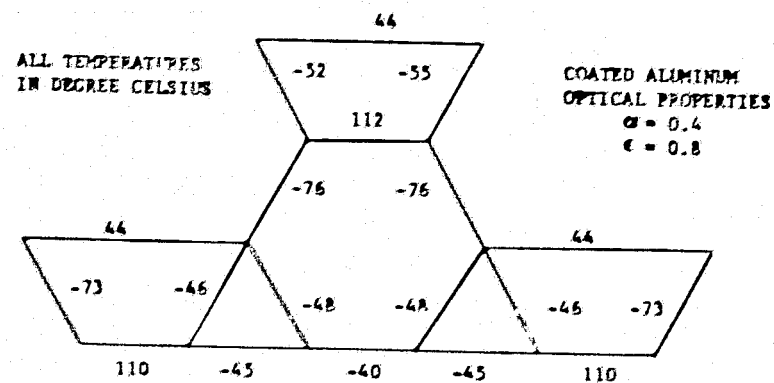
BASIC MODULE TEMPERATURES

This chart shows a portion of the temperatures used for the solar trough members in the computer program analysis. The thermal effects are highly dependent upon the assembly temperature of the elements; however, due to lack of an assembly temperature profile, an assembly temperature of 0°C was used.

BASIC MODULE TEMPERATURES



Graphite Structural Configuration Temperatures



Aluminum Structural Configuration Temperatures

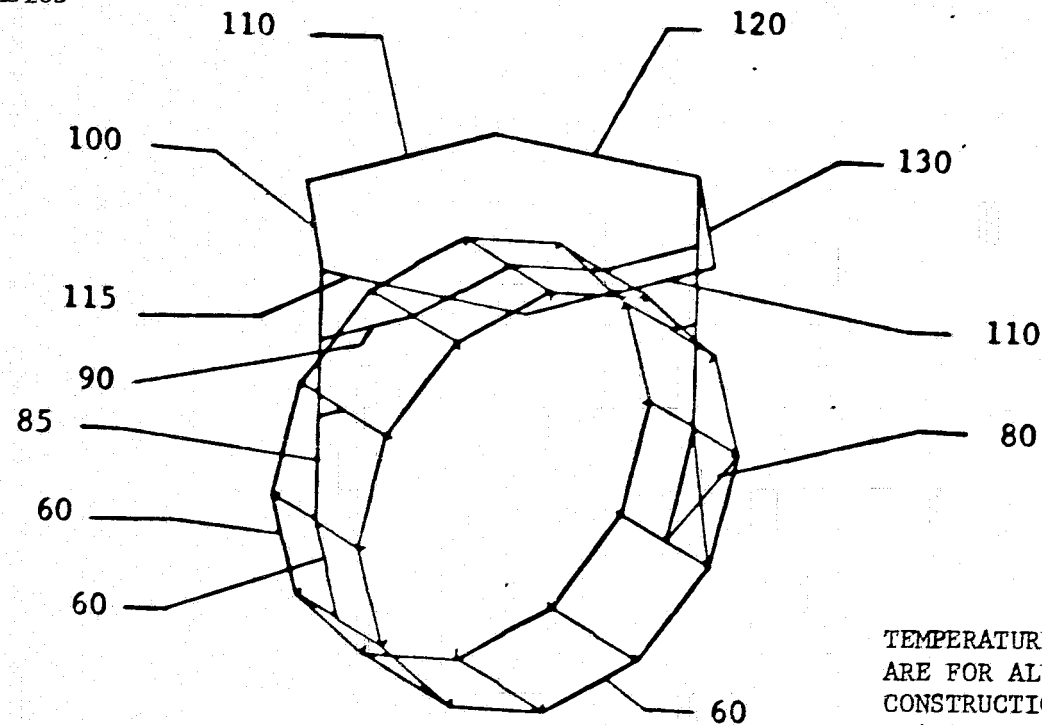
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ANTENNA AND ROTATING RING TEMPERATURES

This chart gives the temperature distribution use for the antenna and rotating ring structure in the computer program analysis.

ANTENNA AND ROTATING RING TEMPERATURES

ALL TEMPERATURES
IN DEGREE CELSIUS



TEMPERATURES SHOWN
ARE FOR ALUMINUM
CONSTRUCTION.
GRAPHITE CONSTRUCTION
APPROX. 3°C HIGHER.

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MATERIAL PROPERTIES

This chart gives a quick summary of the material properties for two temperature values used in the program for aluminum and graphite material. The graphite construction of the elemental beams section assumes 60% fiber volume, 10 mils thick and made up as indicated (0, $\diamond$, 0).

MATERIAL PROPERTIES

GRAPHITE COMPOSITE MATERIAL PROPERTIES

PROPERTY	ROOM TEMPERATURE	177°C (350°F)
ALLOWABLE STRENGTH		
F_{tu}	$4.27 \times 10^8 \text{ N/M}^2$ (62 KSI)	$3.59 \times 10^8 \text{ N/M}^2$ (52 KSI)
F_{ty}	$3.93 \times 10^8 \text{ N/M}^2$ (57 KSI)	$2.28 \times 10^8 \text{ N/M}^2$ (33 KSI)
F_{cy}	$1.58 \times 10^8 \text{ N/M}^2$ (23 KSI)	$1.17 \times 10^8 \text{ N/M}^2$ (17 KSI)
MODULUS OF ELASTICITY		
E_x	$9.65 \times 10^{10} \text{ N/M}^2$ (14 MSI)	$9.52 \times 10^{10} \text{ N/M}^2$ (13.8 MSI)
E_y	$2.76 \times 10^{10} \text{ N/M}^2$ (4 MSI)	$2.41 \times 10^{10} \text{ N/M}^2$ (3.5 MSI)
SHEAR MODULUS		
G_{xy}	$2.41 \times 10^{10} \text{ N/M}^2$ (3.5 MSI)	$2.34 \times 10^{10} \text{ N/M}^2$ (3.4 MSI)
POISSON'S RATIOS		
ν_{xy}	.72	.80
DENSITY	1550 KG/M^3 (.057 LBS/IN ³)	1550 KG/M^3 (.056 LBS/IN ³)
COEFFICIENT OF THERMAL EXPANSION $M/M/^\circ C$	$.18 \times 10^{-6}$	$.18 \times 10^{-6}$
FIBER VOLUME $\frac{v_f}{V}$	.60	.60

ALUMINUM (2014-T6) MATERIAL PROPERTIES

PROPERTY	ROOM TEMPERATURE	150°C (300°F)
ALLOWABLE STRENGTH		
F_{tu}	$4.5 \times 10^8 \text{ N/M}^2$ (65 KSI)	$2.41 \times 10^8 \text{ N/M}^2$ (35 KSI)
F_{ty}	$3.93 \times 10^8 \text{ N/M}^2$ (57 KSI)	$2.0 \times 10^8 \text{ N/M}^2$ (29 KSI)
F_{cy}	$4.00 \times 10^8 \text{ N/M}^2$ (58 KSI)	$1.72 \times 10^8 \text{ N/M}^2$ (25 KSI)
MODULUS OF ELASTICITY	$7.24 \times 10^{10} \text{ N/M}^2$ (10.5 MSI)	$6.83 \times 10^{10} \text{ N/M}^2$ (9.9 MSI)
POISSON'S RATIO	0.33	0.33
DENSITY	2796 KG/M^3 (.101 LBS/IN ³)	2796 KG/M^3 (.101 LBS/IN ³)
COEFFICIENT OF THERMAL EXPANSION	$22.3 \times 10^{-6} \text{ M/M/}^\circ C$ ($12.4 \times 10^{-6} \text{ IN/IN/}^\circ F$)	$23.4 \times 10^{-6} \text{ M/M/}^\circ C$ ($13.0 \times 10^{-6} \text{ IN/IN/}^\circ F$)

TENSION TIE MATERIAL PROPERTIES

PROPERTY	GRAPHITE	SPRING WIRE 302 CRES
ULTIMATE TENSILE STRENGTH	$16.5 \times 10^8 \text{ N/M}^2$ (240 KSI)	$17.6 \times 10^8 \text{ N/M}^2$ (255 KSI)
MODULUS OF ELASTICITY	$2.34 \times 10^{11} \text{ N/M}^2$ (34 MSI)	$1.95 \times 10^{11} \text{ N/M}^2$ (28.2 MSI)
DENSITY	1800 KG/M^3 (.065 LBS/IN ³)	7920 KG/M^3 (.286 LBS/IN ³)
COEFFICIENT OF THERMAL EXPANSION	$-0.4 \times 10^{-6} \text{ M/M/}^\circ C$ ($-0.22 \times 10^{-6} \text{ IN/IN/}^\circ F$)	$14.4 \times 10^{-6} \text{ M/M/}^\circ C$ ($8.0 \times 10^{-6} \text{ IN/IN/}^\circ F$)

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MECHANICAL LOADING

For this study the only mechanical loading considered to be acting on the structure is the reflector pretension loading. Other loadings, such as solar pressure and gravity gradients, are considered negligible for this study. Loading due to solar blankets is not known and therefore not considered.

MECHANICAL LOADING

REFLECTOR LOADING

REFLECTOR MATERIAL:

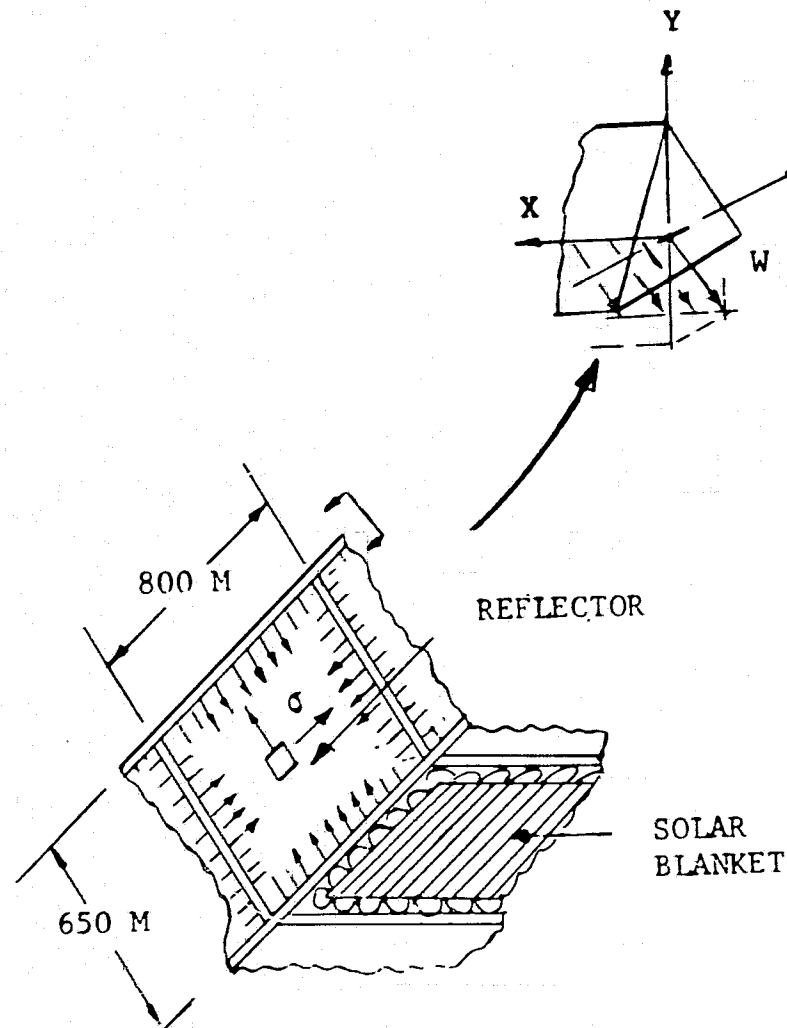
1/2 MIL ALUM. KAPTON
 $t = 1.27 \times 10^{-5} \text{ M}$

REFLECTOR PRETENSION:

$\sigma = 0.517 \text{ MPA (75 PSI)}$

EQUIVALENT LOADING:

$w = \sigma t = 6.567 \text{ N/M (.0375 LB/IN)}$



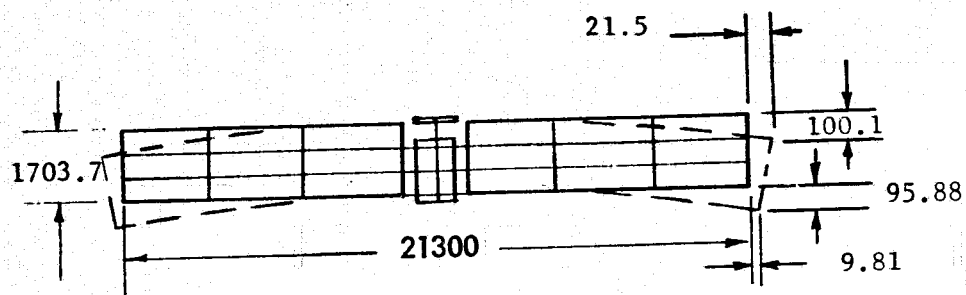
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DEFLECTIONS—SPS UNITS

This chart summarizes the end deflections of the SPS unit for the three analysis cases considered. The pretension, the force necessary to keep the X-braces from becoming compressed under the given thermal and loading environment, is as follows:

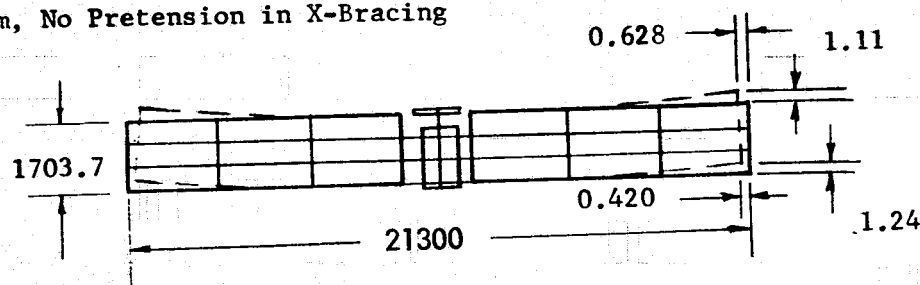
Aluminum design	Steel X-bracing	552 MPa (80 Ksi)
Graphite design	Graphite X-bracing	137 MPa (20 Ksi)

DEFLECTIONS - SPS UNIT

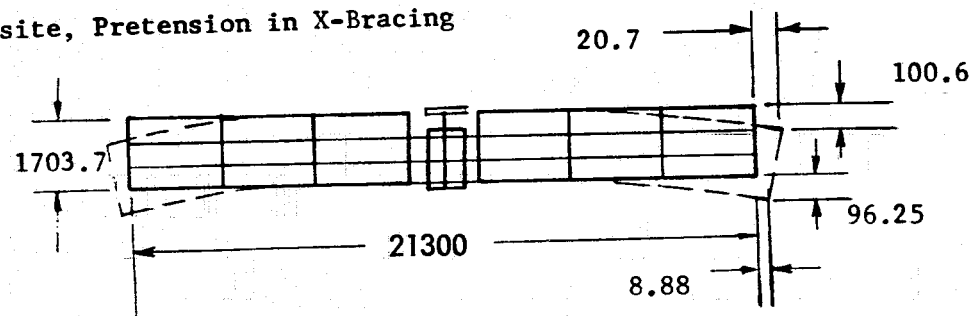


NOTE:
ALL DIMENSIONS
IN METERS

a) Case 1. Aluminum, No Pretension in X-Bracing



b) Case 2. Composite, Pretension in X-Bracing



c) Case 3. Aluminum, Pretension in X-Bracing

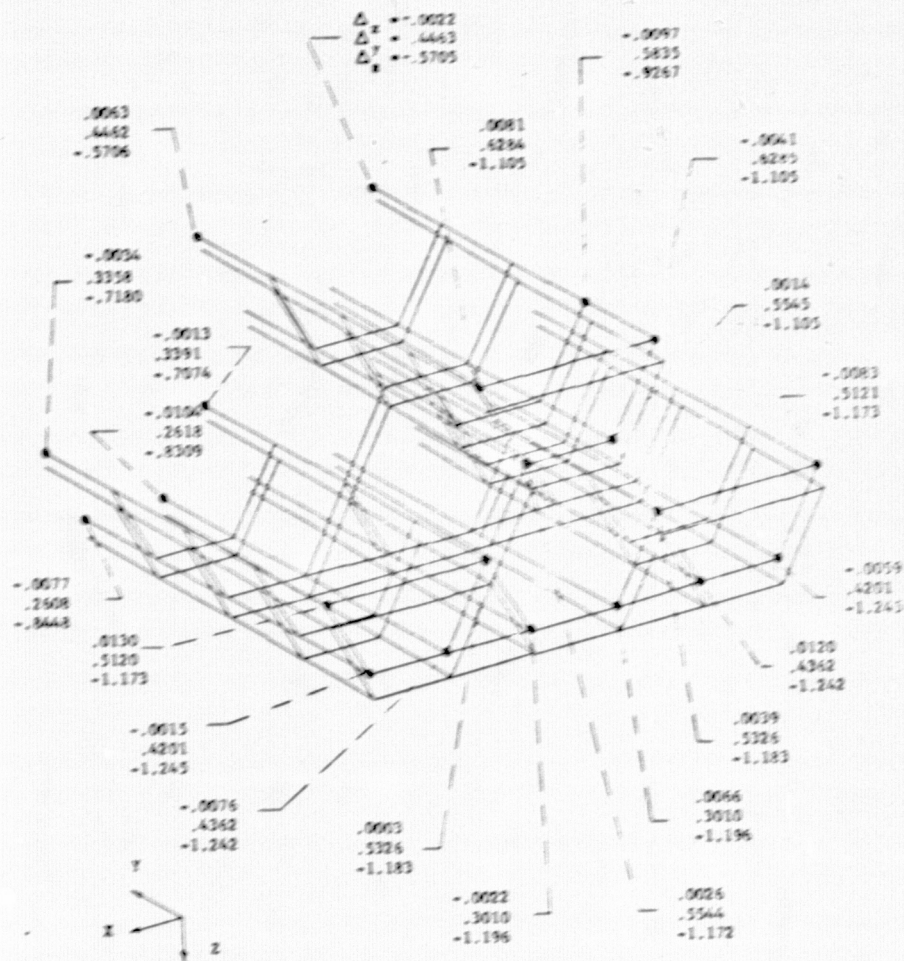
DEFLECTIONS ARE SEVERAL ORDERS
OF MAGNITUDE LARGER FOR
ALUMINUM AS FOR GRAPHITE MATERIAL.

DEFLECTIONS—END MODULE

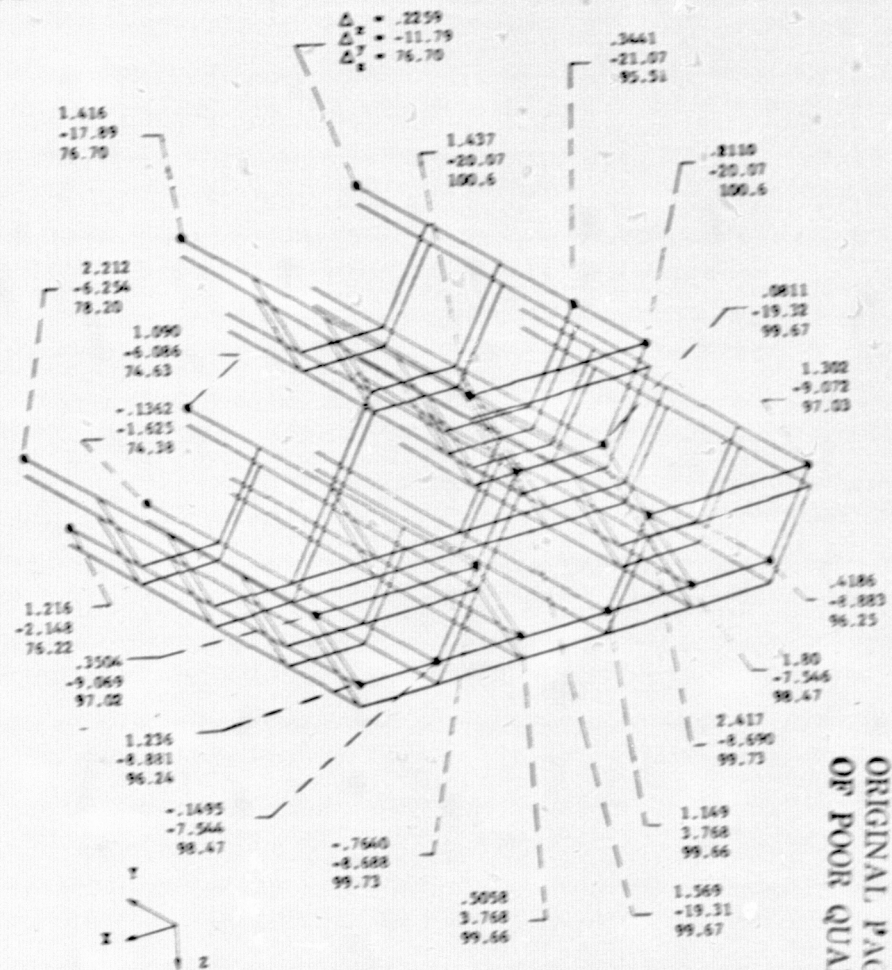
This chart shows the displacements for selected points on the extreme end of the SPS units for the graphite and aluminum material construction under pretensioning. A comparison of the displacements will give the amount of distortion that the reflectors, solar blankets, and other equipment would have to undergo for the various materials. The CRT sketches cannot be compared directly due to the displacement scale factors used by the computer program.

DEFLECTIONS - END MODULE

NOTE:
ALL DIMENSIONS
IN METERS



Substructure No. 1, Case 2, Displacements



Substructure No. 1, Case 3, Displacements

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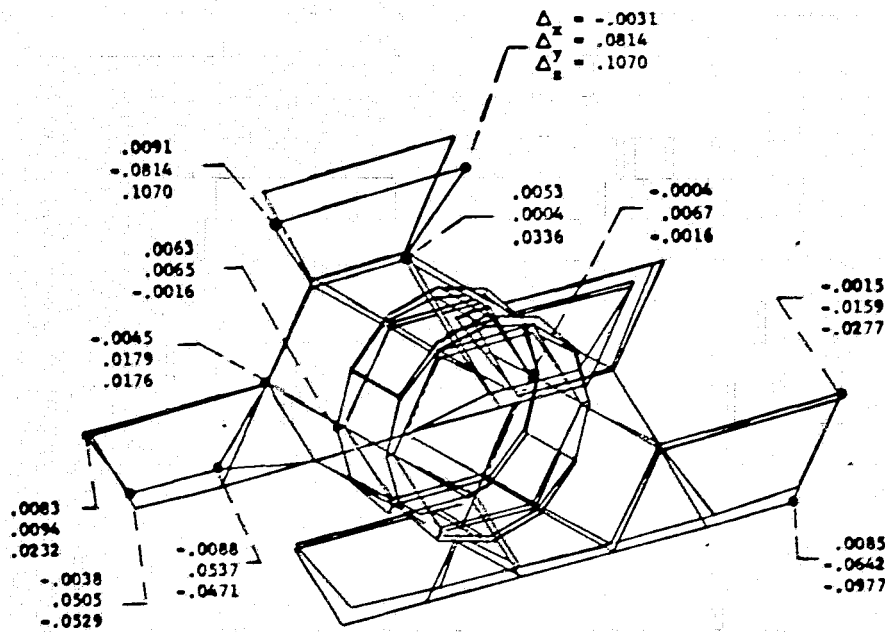
DEFLECTIONS—CENTER STRUCTURE

This chart is similar to the previous one, except the displacements are for the center section.

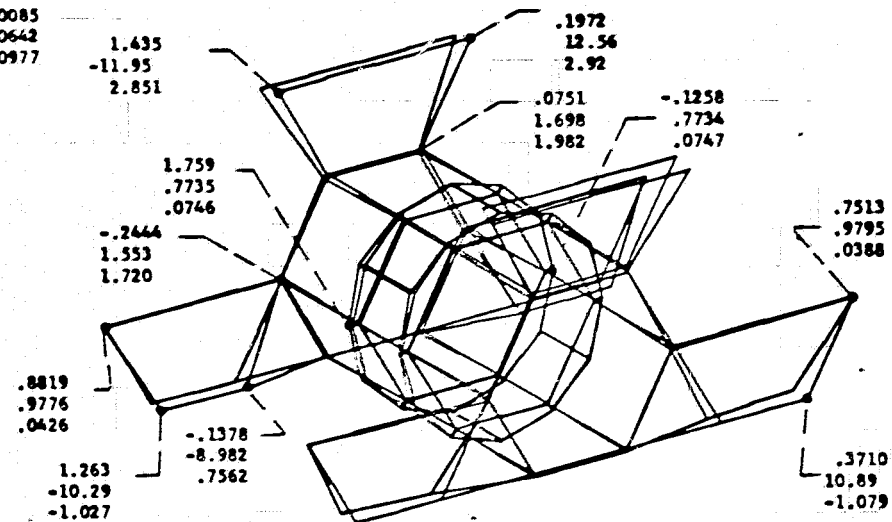
DEFLECTIONS - CENTER STRUCTURE

NOTE:
ALL DIMENSIONS
IN METERS

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a) Case 2, Composite Material



b) Case 3, Aluminum Material



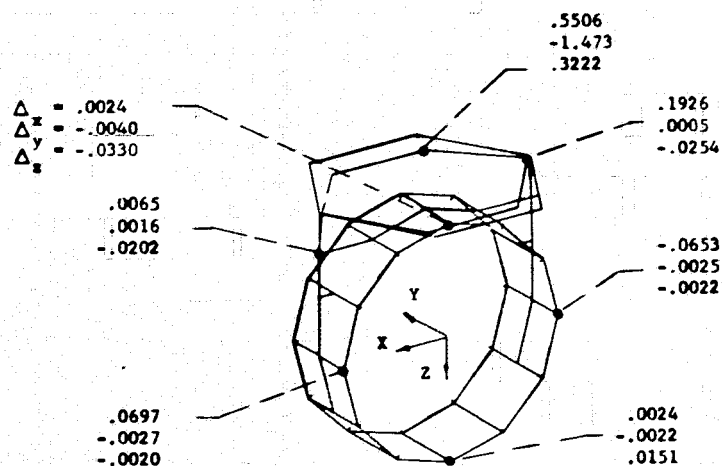
DEFLECTIONS—ANTENNA AND ROTATING RING

This chart is similar to the two previous charts, except the displacements are for the antenna and rotating ring. For this case, the same scale factor was used for the CRT plot of the deformed shapes. A comparison of the two CRT's indicated a considerable distortion of the rotating rings.

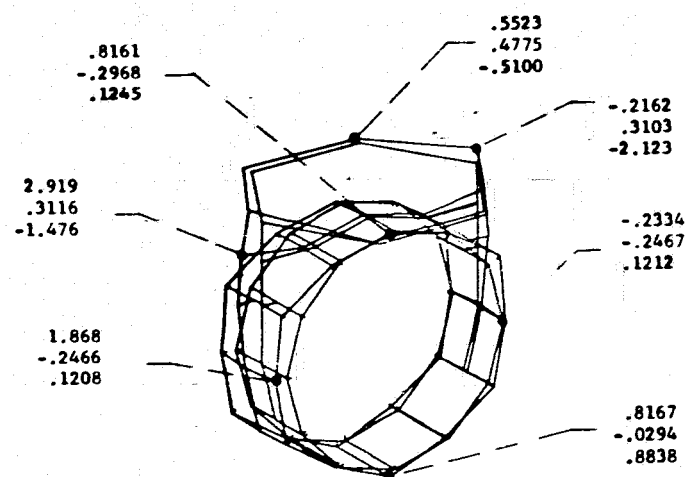
DEFLECTIONS - ANTENNA AND ROTATING RING

NOTE:
ALL DIMENSIONS
IN METERS

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a) Case 2, Composite Material



b) Case 3, Aluminum Material



RESULTS—STRESS

The chart presents the stress of selected members for the graphite and aluminum material construction under pretensioning. As shown in the table, the stresses are much higher for the aluminum case than for the graphite case. The maximum bending stresses for many of the members exceed the crippling allowables of the elemental beam caps.

RESULTS - STRESS

STRESS MAGNITUDES ARE 3 TIMES
AND LARGER FOR ALUMINUM AS
FOR GRAPHITE MATERIAL.

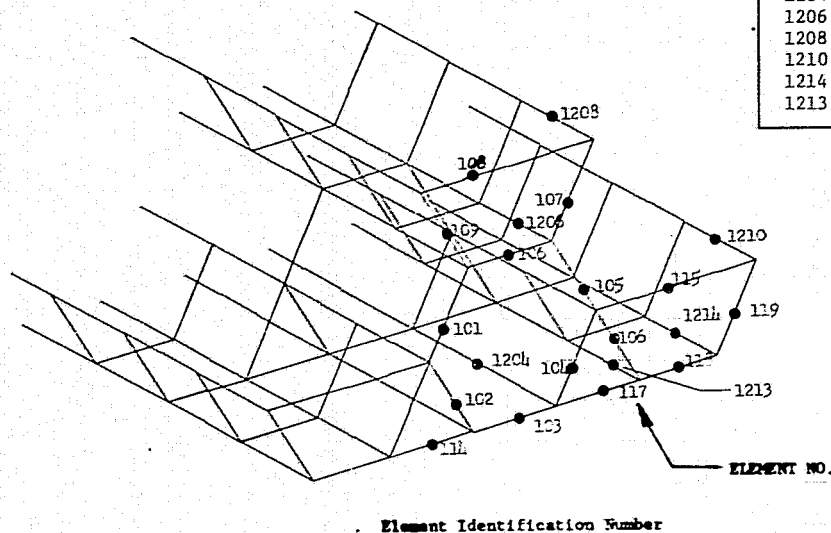


TABLE . COMPRESSIVE STRESS COMPARISON								
ELEMENT NO.	CASE 2, COMPOSITE				CASE 3, ALUMINUM			
	AXIAL STRESS		MAX. BEND STRESS		AXIAL STRESS		MAX. BEND STRESS	
	MPa	(KSI)	MPa	(KSI)	MPa	(KSI)	MPa	(KSI)
101	-1.01	(-.14)	-8.72	(-1.26)	-5.88	(-.85)	-131.7	(-19.10)
102	-1.02	(-.14)	-1.89	(-.27)	-2.85	(-.41)	-58.4	(-8.47)
103	-0.87	(-.12)	-1.47	(-.21)	-9.54	(-1.38)	-72.7	(-10.50)
104	-1.02	(-.14)	-1.89	(-.27)	-2.85	(-.41)	-58.4	(-8.47)
105	-1.02	(-.14)	-8.72	(-1.26)	-5.88	(-.85)	-131.7	(-19.10)
106	-1.99	(-.28)	-4.98	(-.72)	-9.81	(-1.42)	-41.3	(-5.99)
107	-7.68	(-1.11)	-15.90	(-2.31)	-10.31	(-1.50)	-80.2	(-11.63)
108	-0.71	(-.10)	-2.21	(-.32)	-3.20	(-.46)	-8.4	(-1.22)
109	-7.68	(-1.11)	-15.92	(-2.31)	-10.03	(-1.45)	-80.2	(-11.63)
114	-0.21	(-.03)	-1.15	(-.16)	-2.74	(-.39)	-129.5	(-18.78)
115	-0.70	(-.10)	-2.60	(-.37)	-4.46	(-.64)	-22.5	(-3.26)
116	-7.65	(-1.11)	-16.59	(-2.41)	-15.70	(-2.28)	-110.1	(-15.97)
117	-0.21	(-.03)	-1.52	(-.17)	-2.74	(-.39)	-129.5	(-18.78)
118	-1.19	(-.17)	-3.85	(-.55)	-5.02	(-.73)	-59.5	(-8.63)
119	-7.61	(-1.10)	-1.52	(-.22)	-11.06	(-1.68)	-36.1	(-5.24)
1204	-2.08	(-.30)	-2.73	(-.39)	-8.26	(-1.20)	-80.7	(-11.70)
1206	-7.58	(-1.10)	-17.62	(-2.56)	-27.48	(-3.99)	-85.1	(-12.34)
1208	-7.01	(-1.02)	-18.09	(-2.62)	-8.66	(-1.26)	-49.6	(-7.19)
1210	-6.94	(-1.01)	-18.53	(-2.69)	-9.83	(-1.43)	-37.1	(-5.38)
1214	-6.90	(-1.00)	-16.22	(-2.35)	-12.71	(-1.84)	-17.7	(-2.57)
1213	-7.53	(-1.09)	-9.11	(-1.32)	-32.55	(-4.72)	-102.6	(-14.88)

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RESULTS—THERMAL EFFECTS

This chart summarizes the results of the thermal effects versus material selection for the study.

RESULTS - THERMAL EFFECTS

SUMMARY

- DEFLECTIONS ARE SEVERAL ORDERS OF MAGNITUDE LARGER FOR ALUMINUM AS FOR GRAPHITE MATERIAL.
- STRESS MAGNITUDES ARE 3 TIMES AND LARGER FOR ALUMINUM AS FOR GRAPHITE MATERIAL.
- PRESTRESSING OF X-BRACING FOR METAL CONSTRUCTION MUCH HIGHER THAN FOR COMPOSITE CONSTRUCTION:

ALUMINUM DESIGN: STEEL X-BRACING 552 MPa (80 KSI)

GRAPHITE DESIGN: GRAPHITE X-BRACING 137 MPa (20 KSI)

STRUCTURAL ANALYSIS PROGRAM (NASTRAN) DEFICIENCIES

This chart lists several of the deficiencies in the NASTRAN computer program for the analysis of SPS structures and large area space structures (LASS) in general. Investigation into, and the correction of, the deficiencies must be accomplished for detailed stress/dynamic analysis of SPS. Possible corrections may be through the additions to the NASTRAN program and/or through the use of DMAP feature of the NASTRAN program. Other correcting alternatives may be through the use of pre- and post-processor programs used in conjunction with the NASTRAN program.

STRUCTURAL ANALYSIS PROGRAM (NASTRAN) DEFICIENCIES

- SUBELEMENT ANALYSIS OF BEAM ELEMENT NOT PRACTICABLE.
- CONCENTRATED AND/OR DISTRIBUTED LOADINGS CAPABILITY FOR BEAM ELEMENT NOT INCORPORATED IN NASTRAN.
- EQUIVALENT JOINT LOADS USED FOR BEAM ELEMENT LOADING. TIME CONSUMING AND HIGH MANPOWER COST.
- STRESS RECOVERY FOR BEAM ELEMENTS COMPLEX. REQUIRES LABORIOUS ERROR PRONE MANUAL OPERATIONS.
- PROGRAM CANNOT REMOVE TENSION-ONLY MEMBER WHICH BECOMES COMPRESSIVE (NEGATIVE).



STUDY RESULTS

This chart is self-explanatory.

STUDY RESULTS

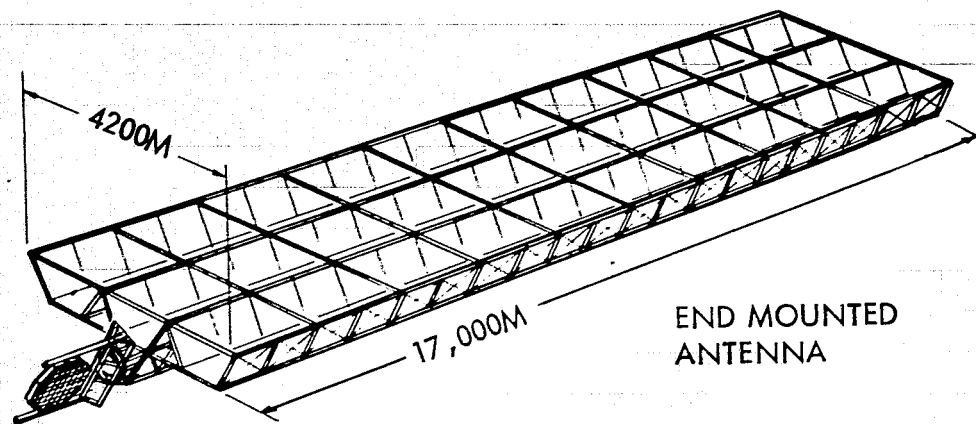
- DEPENDING UPON DESIGN METHODOLOGY AND INGENUITY, THERE WILL BE AN INCREASE IN:
 - ALUMINUM GAUGE SIZE DUE TO CRIPPLING ALLOWABLES
 - STRUCTURAL WEIGHT DUE TO GAUGE SIZE
- THE LARGER DEFLECTIONS DUE TO ALUMINUM CONSTRUCTION MAY PRESENT MORE DESIGN PROBLEMS FOR THE REFLECTORS AND SOLAR BLANKETS ATTACHMENTS, BRUSH DESIGN OF THE ROTATING RING, AND GUIDANCE AND CONTROL REQUIREMENTS
- ADDITIONS TO THE NASTRAN PROGRAM ARE REQUIRED FOR STRESS/DYNAMIC ANALYSIS OF SPS STRUCTURES
- BASED ON THE THERMAL STRESSES AND DEFLECTIONS, ADVANCED COMPOSITES ARE RECOMMENDED FOR SPS STRUCTURES
 - THE COMPOSITES LONG TERM SURVIVABILITY ISSUE MUST BE RESOLVED



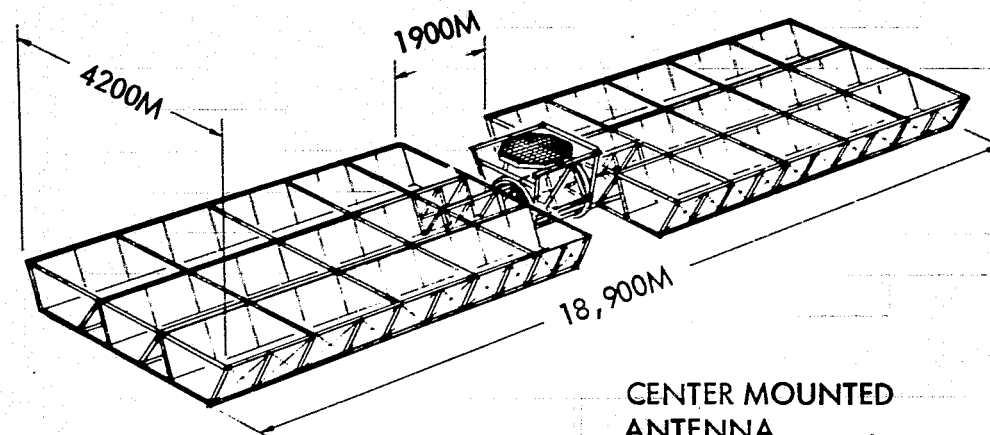
3 TROUGH COPLANAR CONFIGURATIONS

This chart shows the two configurations studied for final point design concept selection.

3 TROUGH CO-PLANAR CONFIGURATIONS



END MOUNTED
ANTENNA



CENTER MOUNTED
ANTENNA

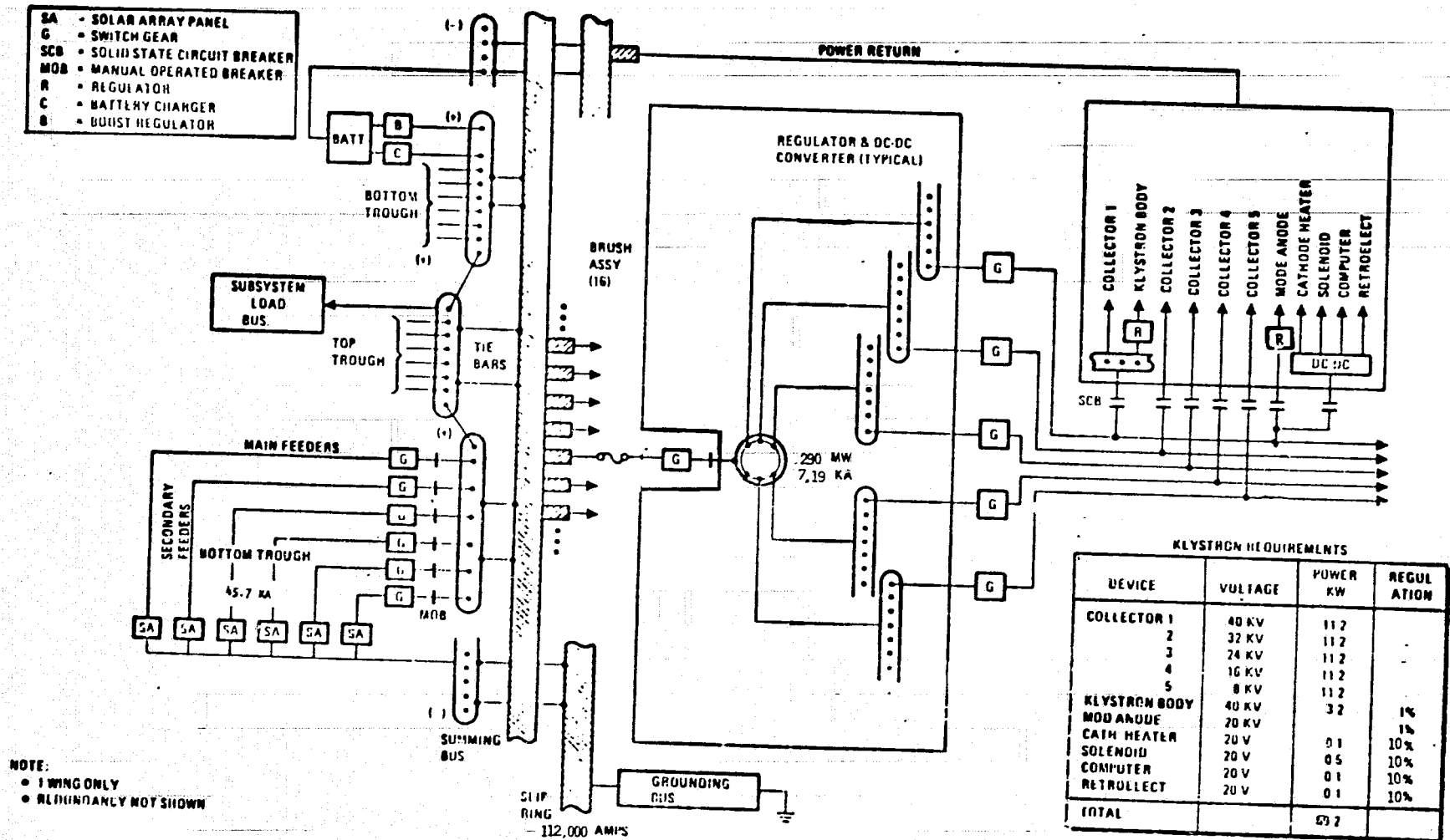
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PDS SIMPLIFIED BLOCK DIAGRAM

This chart illustrates the SPS power distribution interface for the solar photovoltaic concept. The distribution subsystem consists of main feeders, secondary feeders, tie bars, summing buses, voltage converters, regulators, switch gear, slip-rings, brushes and subsystem cabling. The secondary feeders transfer the power from the solar array to the main feeders. The power from main feeders to the summing bus is transferred via switch gears. On-array switching of submodules is used to maintain bus regulation. Mechanical operated breakers (MOB) have been included for safety. The interconnection between slip-rings and the summing buses is performed through tie bars. Individual klystron dc voltage conversion is performed by centralized converters (one for each brush assembly).

The schematic shows one wing of the solar array only. The major difference between configurations is the number of solar panels tied to each summing bus and size of power distribution equipment, i.e., 5 solar panels for the center mounted antenna vs 10 solar panels for the end mounted antenna.

PDS SIMPLIFIED BLOCK DIAGRAM



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SOLAR PHOTOVOLTAIC POWER CONVERSION MASS STATEMENT
~10⁶ kg

The mass of the final two coplanar configurations are shown in the chart. The collector array for the center mounted antenna concept results in 1.636×10⁶ kg less weight due primarily to the difference in power distribution weights, (1.503×10⁶ kg vs 2.889×10⁶ kg). The antenna section for both concepts are taken to be the same. Total SPS dry weight with growth shows a savings of 2.045×10⁶ kg for the center mounted antenna concept (34.714×10⁶ kg vs 36.759×10⁶ kg).

SOLAR PHOTOVOLTAIC POWER CONVERSION MASS STATEMENT - $\sim 10^6$ KG

SUBSYSTEM	CO-PLANAR (3 TROUGH)	
	END MOUNTED ANTENNA	CENTER MOUNTED ANTENNA
COLLECTOR ARRAY		
STRUCTURE AND MECHANISMS	(1.811)	(1.561)
PRIMARY STRUCTURE	.997	1.012
SECONDARY STRUCTURE	.581	.316
MECHANISM	.233	.233
ATTITUDE CONTROL	(0.116)	(0.116)
POWER SOURCE	(8.440)	(8.440)
SOLAR PANELS	7.258	7.258
SOLAR REFLECTORS	1.182	1.182
POWER DISTRIBUTION AND CONTROL	(2.889)	(1.503)
POWER CONDITIONING EQUIPMENT	(.262)	(.262)
POWER DISTRIBUTION	(2.627)	(1.241)
CONDUCTORS AND INSULATION	(2.408)	1.022
SLIP RINGS	.219	.219
INFORMATION MANAGEMENT & CONTROL	(0.050)	(0.050)
DATA PROCESSING	0.021	0.021
INSTRUMENTATION	0.029	0.029
TOTAL ARRAY, DRY	13.306	11.670



SOLAR PHOTOVOLTAIC POWER CONVERSION MASS STATEMENT - $\sim 10^6$ KG (CONT)

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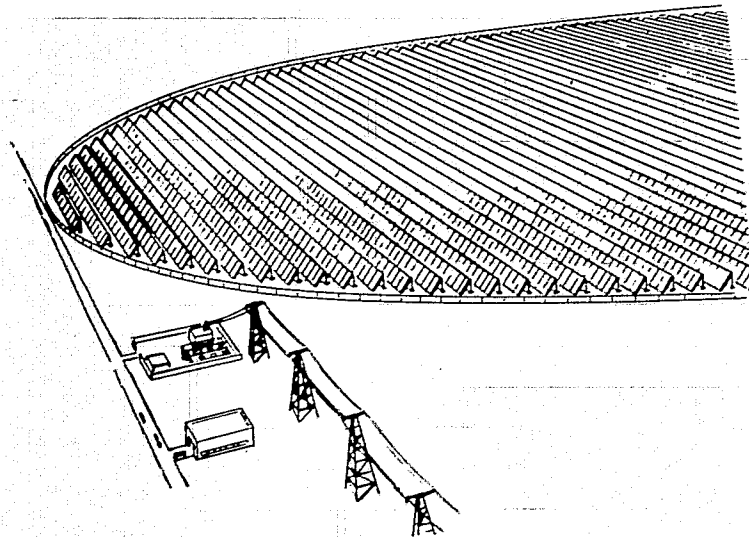
SUBSYSTEM	CO-PLANAR (3 TROUGH)	
	END MOUNTED ANTENNA	CENTER MOUNTED ANTENNA
ANTENNA SECTION		
STRUCTURE & MECHANISM	(1.486)	(1.486)
PRIMARY STRUCTURE	.183	.183
SECONDARY STRUCTURE	.972	.972
ANTENNA	.14	.14
MECHANISM	.191	.191
THERMAL CONTROL	(2.457)	(2.457)
KLYSTRON COOLING	.851	.851
INSULATION	.557	.557
RADIATOR	1.049	1.049
MICROWAVE POWER	(7.012)	(7.012)
KLYSTRONS	4.250	4.250
ATT. SEN. ELECTRONICS & PHASE CONTROL	.142	.142
WAVEGUIDES	2.620	2.620
POWER DISTRIBUTION & CONTROL	(4.516)	(4.516)
POWER CONDITIONING EQUIPMENT	2.466	2.466
POWER DISTRIBUTION	(2.050)	(2.050)
CONDUCTOR & INSULATION	(1.904)	(1.904)
SLIP RING BRUSHES	(.139)	(.139)
INFORMATION MANAGEMENT & CONTROL	(.630)	(.630)
DATA PROCESSING	.380	.380
INSTRUMENTATION	.250	.250
TOTAL ANTENNA SECTION	16.101	16.101
TOTAL SPS DRY	29.407	27.771
GROWTH 30%	7.352	6.943
TOTAL SPS DRY WITH GROWTH	36.759	34.714



LAND REQUIREMENTS

This chart defines the land requirements for the system operating conditions established in the baseline SPS transmission concept. The illustrations footprint for the 23 mW/cm^2 peak incident power density is an ellipse with 10 km by 13 km minor and major axis dimensions respectively. Thus the area enclosed by the basic rectenna is calculated to be approximately 102.1 km^2 . With the additional land required for power conversion to utility compatible forms, and with the addition of safety barriers along the rectenna perimeter the required land purchase increases to a range of 121 to 135 km^2 depending on power distribution complexity.

LAND REQUIREMENTS



- 13 KM (N-S) BY 10 KM (E-W)
- 23 MW/CM² MAXIMUM RADIATION AT CENTER OF RECTENNA
- LESS THAN 1 MW/CM² AT FENCE LINE

RECTENNA SITE (TYPICAL)

BASIC RECTENNA (AT 34° N LATITUDE)

DIMENSIONS: 13 KM (N-S) X 10 KM (E-W)
AREA: 102.1 KM²

CONVERSION AND SWITCHYARD

AREA/CONVERSION STATION: 0.65 - 1.3 KM²
NUMBER STATIONS: 12 - 16
AREA REQUIRED: 7.8 - 20.8 KM²

ADDED FOR SAFETY PERIMETER & MISC. (10%)

11 - 12.3 KM²

TOTAL LAND REQUIREMENTS/RECTENNA SITE

121 - 135 KM²

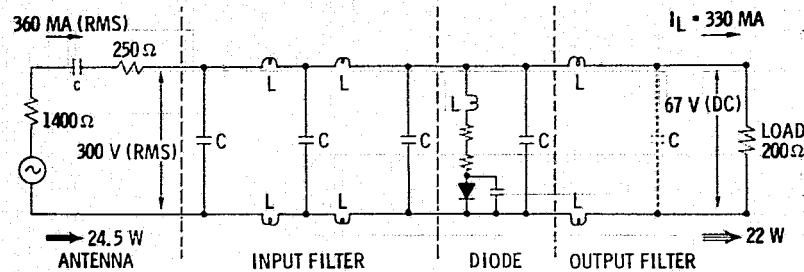
RECTENNA PANELS

This chart illustrates the main physical and electrical parameters of the concept rectenna point design. The panels consist of a multi-layer copper and dielectric panel containing the electrical equivalent of dipole elements, filters and rectifiers. The dimensions used in the point design exercise are shown but it must be understood that these are not necessarily optimum and therefore must be used with caution. The equivalent circuit for a single antenna/rectifier segment is also shown. These segments may be interconnected in series/parallel combinations to provide 13.35 kW "strings" which are considered the smallest switchable elements of the rectenna farm.

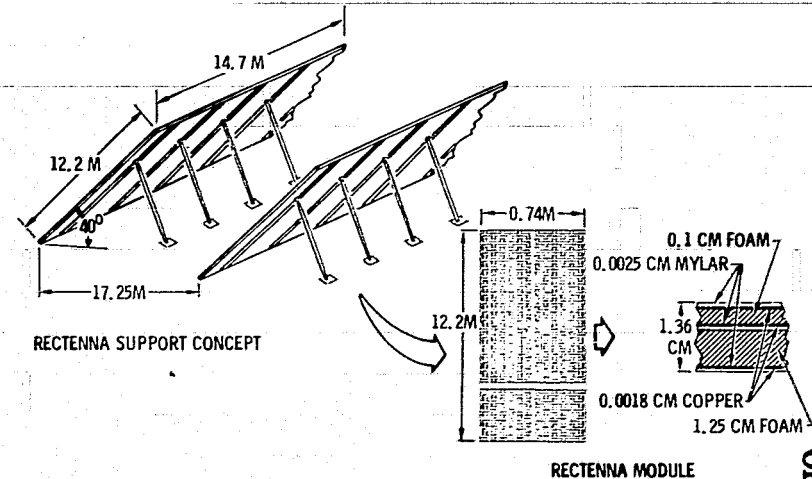
A candidate diode concept is illustrated to show how a GaAs diode may be fabricated and to depict the most desirable approach considering degradation effects.

Baseline design requirements result in a preliminary requirement for more than 390,000 panels, comprising over 369,000 individually switchable power strings. With a basic panel area of $\sim 180 \text{ m}^2$, the total surface area normal to the incident microwave radiation is about 70.2 km^2 . Design estimates indicate that for this concept it will be necessary to fabricate and install approximately 220×10^6 diodes.

RECTENNA PANELS



EQUIV. CIRCUIT

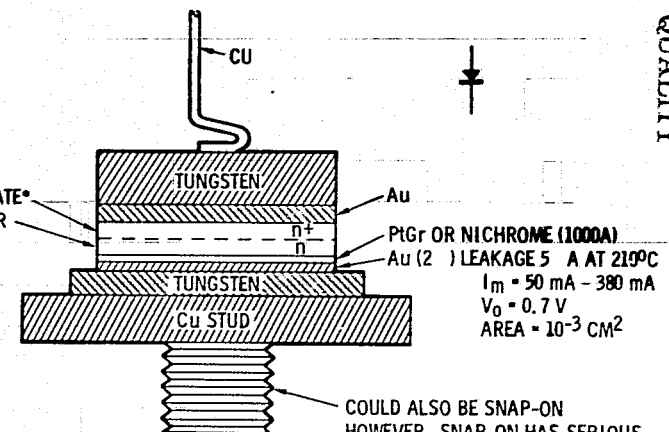


RECTENNA CONSTRUCTION

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STRING	VOLTAGE	~40,000 VOLTS
	CURRENT	0.33 AMPS
	POWER	13.35 KW
	NUMBER	~369,000
PANELS	DIMENSIONS	14.69 x 12.2 M
	AREA/PANEL	179.8 M ²
	NUMBER	390,500
	TOTAL AREA	70.2 KM ²
DIODES	PER STRING	597
	TOTAL	220 x 10 ⁶

(100) GaAs SUBSTRATE*
GaAs EPI LAYER



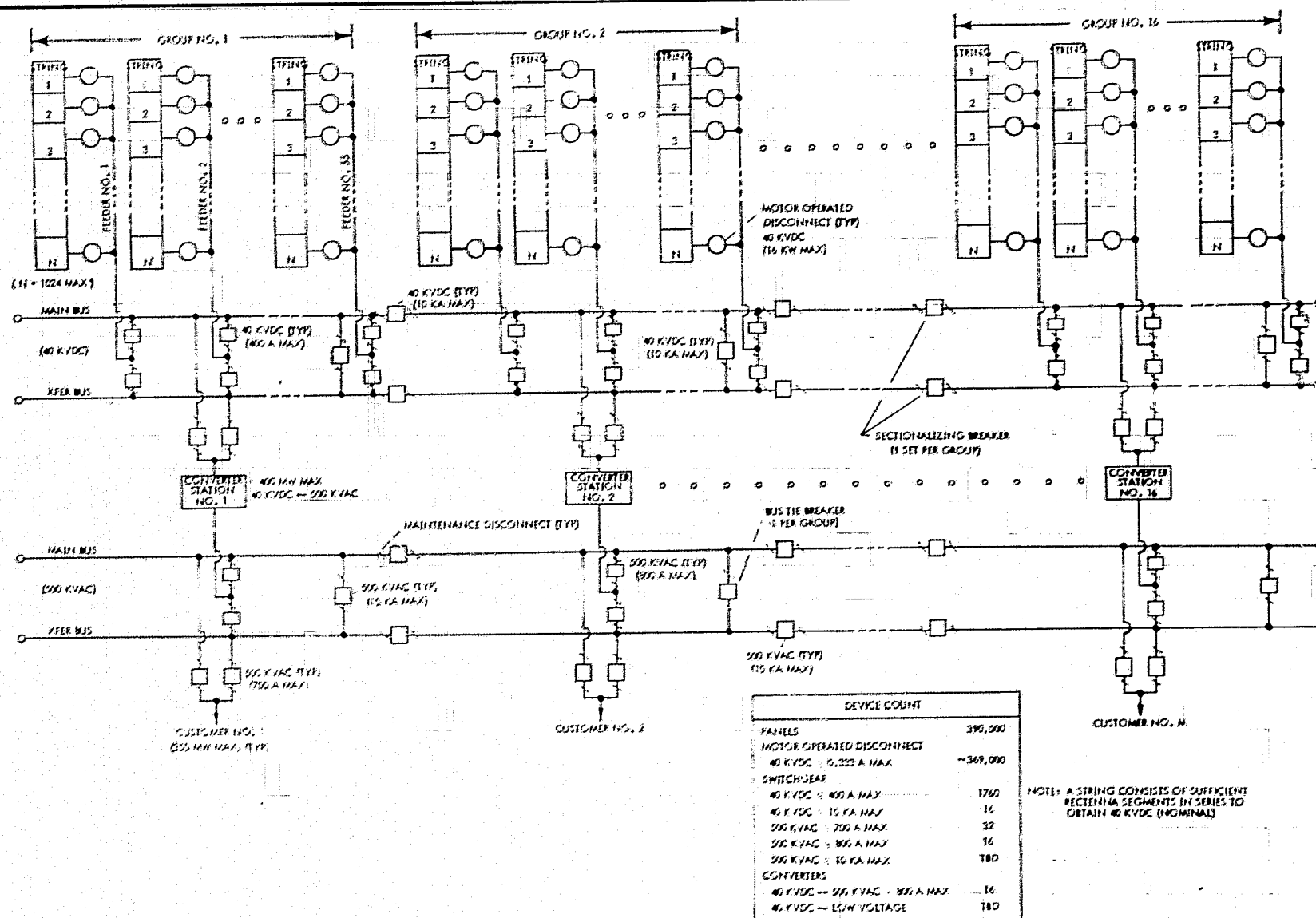
* (111) GaAs IS ALSO A CANDIDATE
FOR SCHOTTKY BARRIER DIODES

DIODE CONCEPT

ELECTRICAL BLOCK DIAGRAM RECTENNA SWITCHYARD

This chart illustrates the design capability of a computer managed conversion/switchyard compatible with the conceptual rectenna point design. Illustrated are the individual rectenna strings, the internal summing (Xfer) busses, the conversion stations, and the main power bus that serves as the interface (with appropriate switchgear) to the various utility networks.

ELECTRICAL BLOCK DIAGRAM RECTENNA SWITCHYARD



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BRIEFING OUTLINE

EXECUTIVE SUMMARY

G. M. HANLEY

REFERENCE CONCEPT CHARACTERISTICS

A. A. NUSSBERGER
R. E. OGLEVIE

REFERENCE CONCEPT CONSTRUCTION

R. E. SEXTON

MICROWAVE TRANSMISSION SYSTEM

G. D. O'CLOCK

TRANSPORTATION SYSTEM

R. P. BERGERON

EXPERIMENT/ VERIFICATION PROGRAM

W. R. RHOTE



ATTITUDE CONTROL AND STATIONKEEPING SUBSYSTEMS
FOR COPLANAR SPS CONFIGURATION

The purpose of the work presented here is to evaluate the coplanar configuration, assess its potential control penalties, develop the ACSS requirements and perform trades to define a new preferred ACSS concept.

ATTITUDE CONTROL AND STATIONKEEPING SUBSYSTEM

FOR

COPLANAR SPS CONFIGURATION

RON OGLEVIE

&

DON CAMILLONE



CONTENTS: ACSS FOR COPLANAR SPS

The subjects to be treated are listed in the chart. Of particular concern is how to accommodate the large unbalanced solar pressure torques inherent in the coplanar configuration. To accomplish this a solar pressure model is developed and its impact on attitude control and stationkeeping is defined. Trades are performed to find ACSS concepts which minimize the RCS propellant and number of thrusters required. A configuration is found which meets the attitude control and stationkeeping requirements with a RCS propellant quantity only slightly greater than required for stationkeeping alone. A gimbaled RCS thruster arrangement is found to substantially reduce the number of RCS thrusters required relative to previous body fixed thruster configurations. The system is capable of significant bank angles (roll) which substantially reduces the collector cosine losses inherent in previous configurations utilizing the Y-POP attitude.

CONTENTS: ACSS FOR COPLANAR SPS

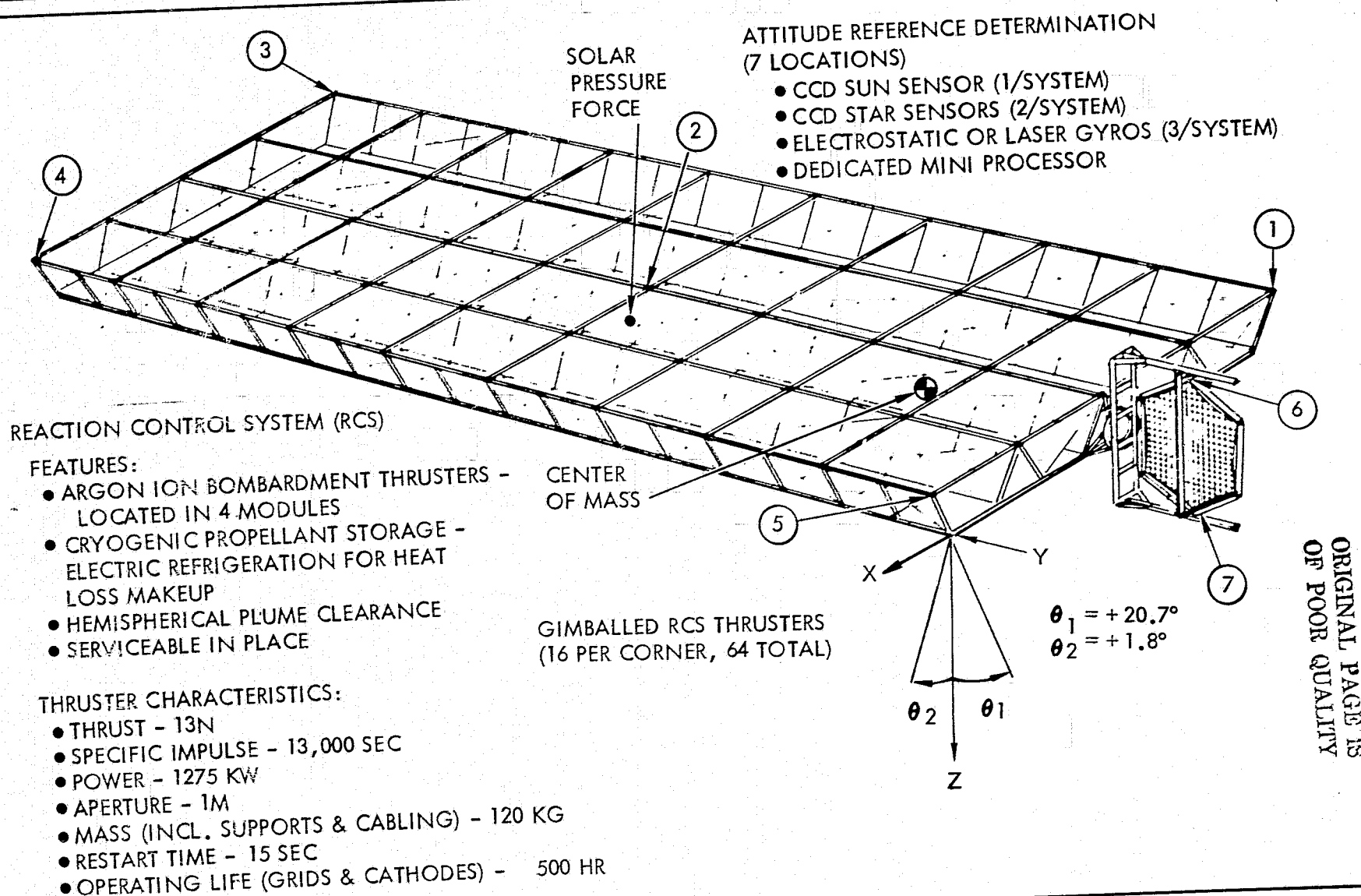
- SOLAR PRESSURE MODEL AND STATIONKEEPING
- IMPACT OF COPLANAR CONFIGURATION ON CONTROL REQUIREMENTS
- SYSTEM TRADES
- BASELINE ACSS DESCRIPTION
- PARTIAL SOLAR POINTING (BANKING) TO REDUCE COLLECTOR COSINE LOSSES
- CONCLUSIONS AND RECOMMENDATIONS



ACSS EQUIPMENT LOCATION

The chart illustrates some of the pertinent features of the new baseline ACSS and shows the location of the subsystems. The RCS subsystem features an argon ion bombardment thruster whose characteristics are enumerated on the figure. The system operates an average of approximately 36 thrusters. A total of 64 thrusters are included to provide the required redundancy. The redundancy was based on an annual maintenance interval, 5000 hour thruster grid life time and 5 year thruster MTBF. Sixteen thrusters are located on the lower portion of each corner of the collector. Each thruster is gimballed individually to facilitate thruster servicing to permit operation of adjacent thrusters during servicing, and to provide redundancy. The thrusters nominally provide a force approximately in the direction of the sun to counter the solar pressure force (stationkeeping) which is the dominant thruster requirement. The thrusters are gimballed through small angles (as illustrated) and differentially throttled to provide the remaining forces and torques for stationkeeping and attitude control. Also given in the chart are the location and type of sensors that make up the Attitude Reference Determination System (ARDS). The total mass of the ACSS with propellant is 108.5×10^3 kg and the average operating power is 33.87 megawatts.

ACSS EQUIPMENT LOCATIONS

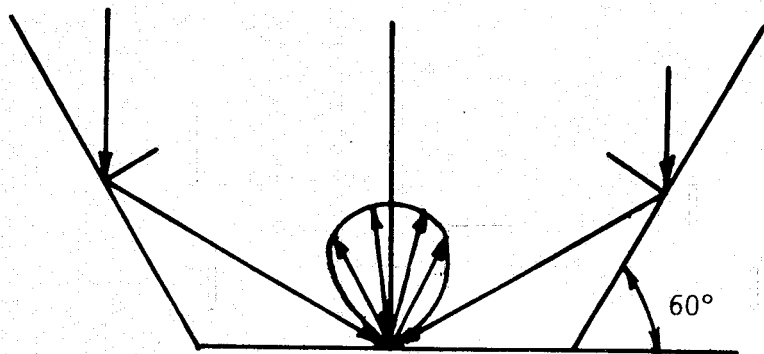


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SOLAR PRESSURE FORCE MODEL

Because of the impact of the solar pressure force on the coplanar design, a more accurate solar pressure force model was developed. To derive the expressions to describe the forces, the assumption was made that energy was reflected specularly off the reflectors and diffusely off the blankets. A complete description of the solar pressure force would have to include the contribution of the force due to the reflections from the blanket back onto the reflectors. However, this contribution is neglected because it is negligible compared to the other forces. The chart shows the major steps that were taken to obtain the final force expression as a function of the capture area of the blanket.

SOLAR PRESSURE FORCE MODEL



A_b = TOTAL AREA OF BLANKET

A_c = CAPTURE AREA OF BLANKET = $CR \times A_b$

A_R = TOTAL AREA OF REFLECTOR

R_b = REFLECTIVITY COEFFICIENT OF BLANKET = 0.17

R_R = REFLECTIVITY COEFFICIENT OF REFLECTOR = 0.90

$$\Sigma F_{sp} = \underbrace{P_s A_b (1 + R_b)}_{\text{FORCE ON BLANKET FROM DIRECT SUNLIGHT}} + \underbrace{P_s A_R \cos 60^\circ [(1 - R_R) \sin^2 60^\circ + (1 + R_R) \cos^2 60^\circ]}_{\text{FORCE ON REFLECTORS FROM DIRECT SUNLIGHT}} + \underbrace{2(1 + R_b) R_R P_s A_b \cos^2 60^\circ}_{\text{FORCE ON BLANKET FROM REFLECTED SUNLIGHT}}$$

+ REFLECTIONS FROM BLANKET BACK ON REFLECTORS (NEGLECTED)

$$\Sigma F_{sp} = P_s [A_b (1 + R_b) (1 + R_R / 2) + A_R / 2 (1 - R_R / 2)]$$

FOR $CR = 2$, $A_c = 2A_b$ AND SUBSTITUTING $R_b = 0.17$ & $R_R = 0.9$

$$\Sigma F_{sp} = 1.125 P_s A_c$$

OVERALL COLLECTOR HAS EQUIVALENT REFLECTIVITY = 0.125

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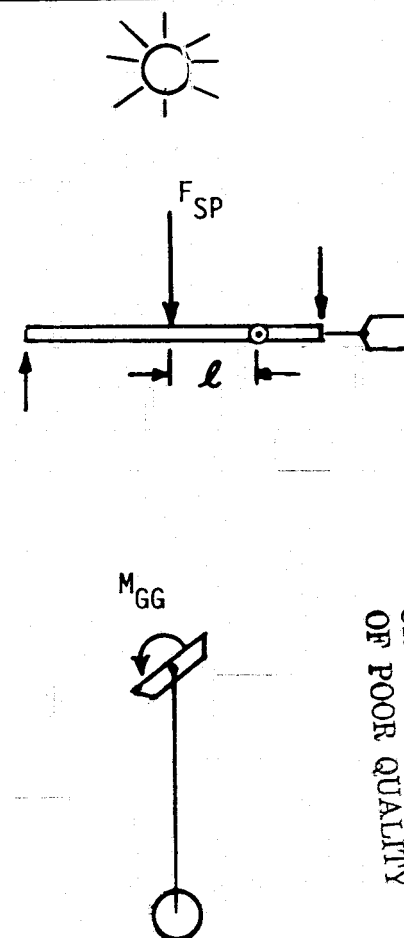
ATTITUDE CONTROL RCS REQUIREMENTS

The SPS coplanar configuration results in substantial solar pressure and gravity gradient torque. The solar pressure torque is due to the end-mounted MW antenna causing a large center of mass offset relative to the center of pressure. The relatively flat design of the spacecraft results in large cyclical gravity gradient torque about the axis perpendicular to the orbit plane (negligible inertia balancing). The S/C orientation is Y-POP; where the Y-axis is the long axis of vehicle.

These torques were the dominant factors driving the RCS propellant and thruster requirements. The effect of these factors on the RCS requirements for attitude control are shown in the chart. The data assumes the disturbance torques are balanced with RCS torque couples and are obtained independently of the thrusters providing the stationkeeping forces.

ATTITUDE CONTROL RCS REQUIREMENTS**

FUNCTION	PROPELLANT MASS (% S/C MASS OVER 30 YEARS)	NO. OF THRUSTERS*		
		X	Y	Z
• SOLAR PRESSURE • ANTENNA RADIATION PRESSURE	4.01			15.2
• GRAVITY GRADIENT • ABOUT X • ABOUT Y • ABOUT Z	0.124 1.38 0.124	1.6		16.4
	1.62	1.6		18.1
TOTALS	5.63	1.6		33.3



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*TORQUE COUPLES ASSUMED

**100-CM ARGON THRUSTERS, $T = 13n$, $I_{SP} = 13,000 \text{ SEC}$, $M_{S/C} = 36.6 \times 10^6 \text{ KG}$, $I_{XX} = 9.86 \times 10^{14} \text{ KG-M}^2$,
 $I_{YY} = 3.09 \times 10^{12} \text{ KG-M}^2$, $I_{ZZ} = 10.2 \times 10^{14} \text{ KG-M}^2$
 (INCLUDING GROWTH)

STATIONKEEPING RCS REQUIREMENTS

For small contemporary spacecraft the dominant stationkeeping ΔV is to correct the solar - lunar perturbations. However, the chart clearly shows that for the SPS vehicle the solar pressure perturbation dominates by a factor greater than five. The contribution of this force on the stationkeeping requirements is approximately 85% of the propellant mass and 57% of the thrusters.

STATIONKEEPING RCS REQUIREMENTS ☆

FUNCTION	ΔV (m/s/YR)	PROPELLANT MASS (% S/C MASS OVER 30 YRS)	THRUST REQUIRED (N)	NUMBER OF THRUSTERS
• SOLAR PRESSURE (E-W) • ANTENNA RADIATION PRESSURE (E-W) • EARTH TRIAXIALITY (E-W) • STATION CHANGE (E-W)	282.5	6.65	328	25.2
• SOLAR -LUNAR PERTURBATION (N-S)	53.3	1.25	124☆☆	19.1☆☆
TOTALS	335.8	7.90	452	44.3

☆ - 100 CM ARGON THRUSTER, T = 13 N, ISP = 13,000 SEC

☆☆ - THRUST ON 50% OF TIME



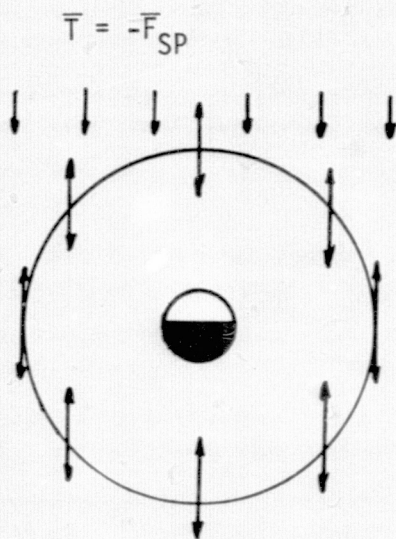
SOLAR PRESSURE STATIONKEEPING PERTURBATIONS AND CORRECTION POLICIES

The chart shows the expressions for the propellant consumption of two solar pressure correction policies, constant cancellation of solar pressure force and Hohmann transfer. The Hohmann transfer correction policy shows a potential propellant savings of 25 percent. However, this savings is not realized because during the coast periods large propellant consumption is required for attitude control which was shown previously. The preferred scheme is the continuous thrusting correction policy, which virtually eliminates the propellant requirements for attitude control due to solar pressure and gravity gradient torques.

SOLAR PRESSURE STATIONKEEPING PERTURBATION AND CORRECTION POLICIES

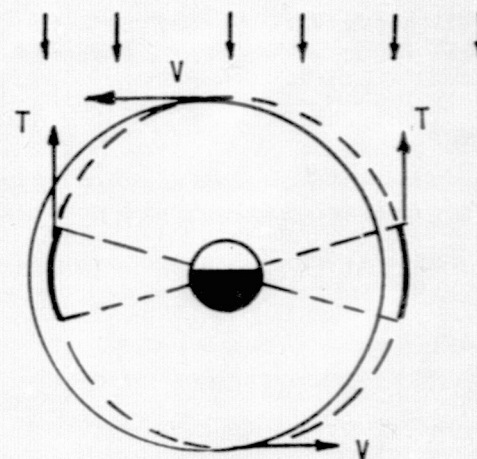
- UNCORRECTED LONGITUDE PERTURBATION = $\Delta\lambda = \pm 2.52^\circ$
- PROBABLY NOT ACCEPTABLE WITH POST-2000 TRAFFIC IN GEO
- CORRECTION POLICIES:

CANCEL SOLAR PRESSURE FORCE



$$M_P = \frac{F_{SP} \Delta t}{I_{SP}}$$

CORRECT AT APOGEE AND PERIAPSIS



$$M_P = \left(\frac{3}{4}\right) \frac{F_{SP} \Delta t}{I_{SP}}$$

(PROPELLANT SAVINGS)

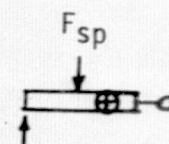
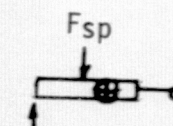
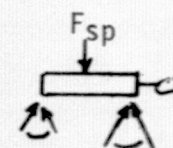
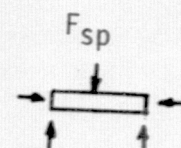
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RCS REQUIREMENTS FOR ALTERNATIVE CONTROL APPROACHES

The chart presents the RCS propellant and thruster requirements for five possible cases including body mounted and gimbale thruster configurations, with and without complete correction of the solar pressure orbit perturbation, and with the addition of two dimensional inertia balancing to the spacecraft or use of the quasi-inertial free drift attitude mode to minimize the impact of the large cyclical gravity gradient torque. From the orbit perturbation considerations, solar pressure stationkeeping correction is required and the preferred correction policy is a continuous force equal and opposite to the solar pressure force. The QI-FDAM approach with its $\pm 18^\circ$ attitude excursions is applicable to a CR=1 configuration but is not appropriate to the current CR=2 design. Inertia balancing is not easily achieved in the present coplanar design concept. Based on these considerations all the alternatives except in the first two rows are rejected. The gimbale thruster configuration is seen to result in a small propellant savings and substantially less thrusters and is selected on that basis.

RCS REQUIREMENTS FOR ALTERNATIVE CONTROL APPROACHES

SYSTEM	THRUSTER CONFIGURATION		PROPELLANT MASS (% S/C MASS) (OVER 30 YR)	NO. OF THRUSTERS REQUIRED*			
	BODY FIXED	GIMBALED		X	Y	Z	TOTAL
COMPLETE STATIONKEEP- ING & ATTITUDE CONTROL	X		8.0	1.6	19.1	34.3	55.0
SAME		X	7.0			35.6	35.6
SAME BUT SOLAR PRESS. TRAJ. PERTURBATION NOT CORRECTED (TORQUE COUPLES FOR ATTITUDE CONTROL)	X		5.4	1.6	19.1	24.3	45.0
SAME BUT SOLAR PRESS. TRAJ. PARTIALLY CORRECTED (TORQUE COUPLES NOT REQ'D)	X		3.9	1.6	19.1	18.3	39.0
SAME BUT QIFDAM OR INERTIA BALANCING EMPLOYED (ELIMIN- ATES CYCLICAL G.G. ABOUT Y-AXIS)	X		8.0	1.6	19.1	26.1	46.8
*DOES NOT INCLUDE REDUNDANCY OR SPARES							



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ACSS MASS SUMMARY

The chart summarizes the mass properties of the ACSS and includes the mass of the individual elements. The total mass is given with and without the propellant. For further information on the equipment locations and pertinent characteristics the reader is referred to an earlier chart entitled "ACSS Equipment Locations".

ACSS MASS SUMMARY

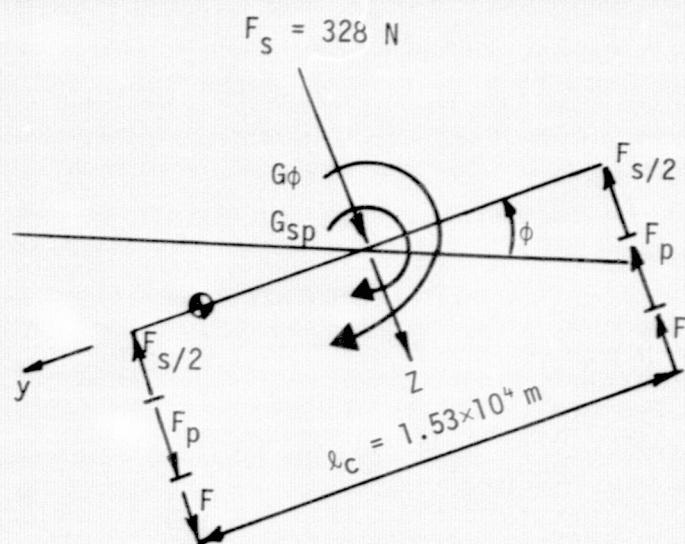
ITEM	MASS (x 10 ⁺³ KG)
ATTITUDE REFERENCE DETERMINATION SYSTEMS (7)	0.32
THRUSTERS - INCLUDING SUPPORT STRUCTURE, 64 @ 120 KG/THRUSTER	7.68
THRUSTER GIMBALS AND MOUNTING	TBD
TANKS, LINES, REFRIGERATION	15.07
POWER PROCESSING EQUIPMENT	TBD
ARGON PROPELLANT - ANNUAL REQUIREMENT	85.39
TOTAL (DRY)	23.07 *
TOTAL (WITH PROPELLANT)	108.46 *

* - NOT INCLUDING TBD ITEMS

MAXIMUM BANK ANGLE

The SPS spacecraft is nominally designed to operate in a Y-POP orientation (i.e., long axis, Y-axis, perpendicular to orbit plane). Major advantages can be realized if the spacecraft is pointed more toward the sun. The collector cosine losses would be reduced which reduce the size of the solar collector, thus a reduction in spacecraft mass and cost. The ideal bank angle is 23.5 degrees which is achievable with an increase in propellant and thrusters. However, partial solar pointing with no propellant and thruster penalty is an advantageous feature. As illustrated in the chart, a roll angle of 9.0 degrees can be realized with no propellant penalty or loss of control authority. Essentially this is achievable by simultaneously countering the large gravity torques and the solar pressure forces through use of thrusters that are predominantly at one end of the S/C. With this bank angle the collector cosine loss is reduced to 3.2% from 8.2% for the Y-POP operation (net savings equal 5%). The chart shows the steps and calculations involved in deriving the bank angle.

MAXIMUM BANK ANGLE



$G_{sp} \triangleq$ SOLAR PRESSURE TORQUE

$G_{\phi} \triangleq$ GG TORQUE DUE TO BANK ANGLE ϕ
 $= \frac{3}{2} W_0^2 (I_Z - I_Y) \sin 2\phi$

$F_{s/2} \triangleq$ FORCE THAT CANCELS OUT SOLAR PRESSURE & TORQUE

$F_p \triangleq$ FORCE TO CONTROL GG TORQUE DUE TO $.30^\circ$ PRINCIPAL AXIS MISALIGNMENT = 5.4 N

$F \triangleq$ FORCE TO CONTROL G_{ϕ}

FROM PREVIOUS ANALYSIS ALL TORQUES EXCEPT G_{ϕ} ARE ACCOUNTED TO INSURE MODULATION $\Sigma F \geq 0$; $F_{s/2} - F_p - F = 0$

$$F = F_{s/2} - F_p = 164 - 5.4 = 158.6 \text{ N}$$

F IS THE MAXIMUM FORCE AVAILABLE TO ROLL VEHICLE BY ANGLE ϕ

$$F l_c = \frac{3}{2} W_0^2 (I_Z - I_Y) \sin 2\phi \text{ AND } \phi = \frac{1}{2} \sin^{-1} \left[\frac{F l_c}{\frac{3}{2} W_0^2 (I_Z - I_Y)} \right]$$

WITH THE VALUES OF F, l_c AND $\frac{3}{2} W_0^2 (I_Z - I_Y) = 7.84 \times 10^6 \text{ NM}$

$$\phi_{\text{MAX}} = 9.02^\circ$$

$$\% \text{ COS LOSS} = 3.16$$

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CONCLUSIONS AND RECOMMENDATIONS

A technique to control the large solar pressure and gravity gradient disturbance torques with minimum penalty in RCS propellant has been developed. The control utilizes gimbale thrusters to provide forces for the stationkeeping and attitude control torques. This combination of gimbale thrusters and dual thruster force operation results in a reduction in RCS propellant and number of thrusters required relative to the previous attitude control and stationkeeping designs. Also the system is capable of small bank angles with no propellant penalties thereby realizing 5% reduction in cosine losses relative to Y-POP operation.

CONCLUSIONS AND RECOMMENDATIONS

- COPLANAR CONFIGURATION HAS LARGE DISTURBANCE TORQUES (SOLAR PRESSURE AND GRAVITY GRADIENT)
- IF SOLAR PRESSURE STATIONKEEPING CORRECTIONS NOT REQUIRED PROPELLANT PENALTIES ARE LARGE AND ALTERNATIVE CONFIGURATIONS SHOULD BE CONSIDERED
- IF SOLAR PRESSURE STATIONKEEPING CORRECTIONS ARE REQUIRED PROPELLANT PENALTIES FOR ATTITUDE CONTROL ARE VIRTUALLY ELIMINATED, (I.E. COPLANAR SPS ACCEPTABLE)
- GIMBALLED THRUSTER ARRANGEMENT DEVELOPED, RESULTS IN FEWER THRUSTERS REQUIRED AND PROPELLANT REDUCTION (RELATIVE TO EARLIER BODY MOUNTED THRUSTERS)
- SMALL ROLL ANGLES CAN BE FLOWN, RESULTS IN SUBSTANTIAL REDUCTION IN COLLECTOR COSINE LOSSES



BRIEFING OUTLINE

EXECUTIVE SUMMARY

G. M. HANLEY

REFERENCE CONCEPT CHARACTERISTICS

A. A. NUSSBERGER
R. E. OGLEVIE

REFERENCE CONCEPT CONSTRUCTION

R. E. SEXTON

MICROWAVE TRANSMISSION SYSTEM

G. D. O'CLOCK

TRANSPORTATION SYSTEM

R. P. BERGERON

EXPERIMENT/ VERIFICATION PROGRAM

W. R. RHOTE

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COPLANAR SATELLITE CONSTRUCTABILITY



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171

170
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CONFIGURATION DESIGN CRITERIA

GROUND RULES

- COPLANAR CONFIGURATION
- 5 GIGAWATT SYSTEM
- END-MOUNTED ANTENNA
- COMPOSITE STRUCTURE
- GEO CONSTRUCTION, 2 SATELLITES PER YEAR
- ELECTRIC COTV
- GaAs SOLAR CELLS, CR = 2

GUIDELINES

- RETAIN DESIREABLE FEATURES OF JUNE '78 DESIGN
 - GEOMETRY
 - CONSTRUCTION CONCEPT

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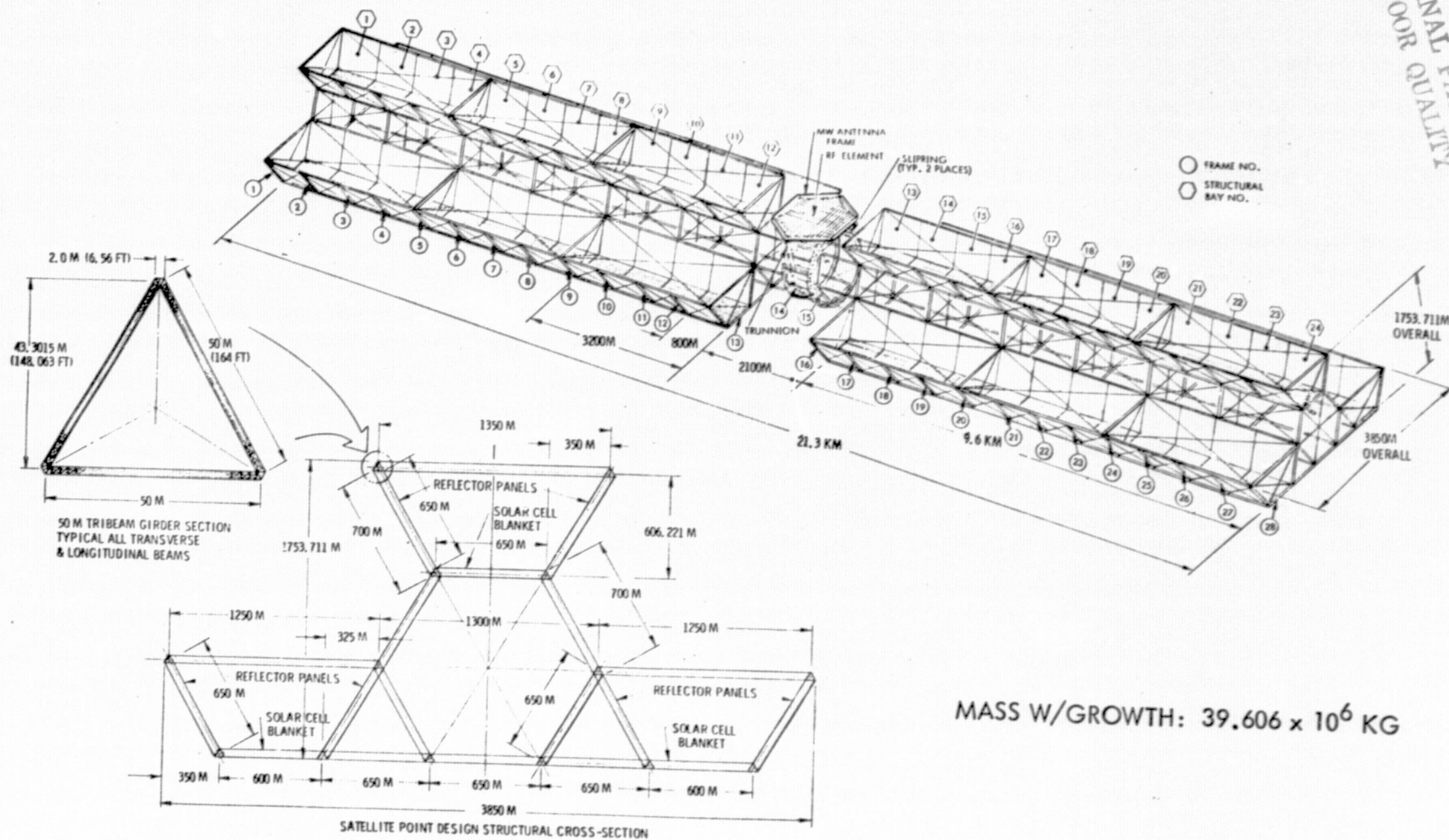
ROCKWELL BASELINE SATELLITE DESCRIPTION, JUNE 1978

The satellite is comprised of two wings and a center section upon which the slip ring and antenna is mounted. Each wing consists of 12 bays 800 m long, numbered as shown on the isometric representation of the satellite. Referring to the cross section view, solar blanket strips 750 m long and 25 m wide are installed along the bottom of the three troughs as indicated, while the reflector panels are installed in the trough sides.

The satellite structure is constructed from 50 m tribeams, the longerons being continuous structure, which are fabricated from the basic building block of 2 m tribeams as shown in the tribeam cross section. The overall construction concept entails use of a satellite construction base, described in a later chart, to construct one wing, commencing with bay 1, complete the center section, and then fabricate the second wing, commencing with Bay 24. Installation of the solar power converter and power distribution equipment is accomplished concurrently with fabrication of each bay.

ROCKWELL BASELINE SATELLITE DESCRIPTION, JUNE 1978

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MASS W/GROWTH: 39.606×10^6 KG

CONSTRUCTION SCHEDULE FOR ROCKWELL BASELINE SATELLITE,
JUNE 1978

A single integrated construction facility builds the structure, installs the solar blankets, the reflectors, the power distribution system and other subsystem elements located in the wings. Construction starts with one wing tip and progresses toward the center section where the rotating joint for the MW antenna is to be located, and hence continues outboard building wing No. 2 and terminating at the wing tip.

The first eight days are designated for preparation of the construction facility, including distribution and installation into dispensers of material (e.g., structure cassettes, solar blankets, etc.) required to commence construction. During this time satellite materials are arriving from LEO daily with delivery scheduled for completion by the 60th day.

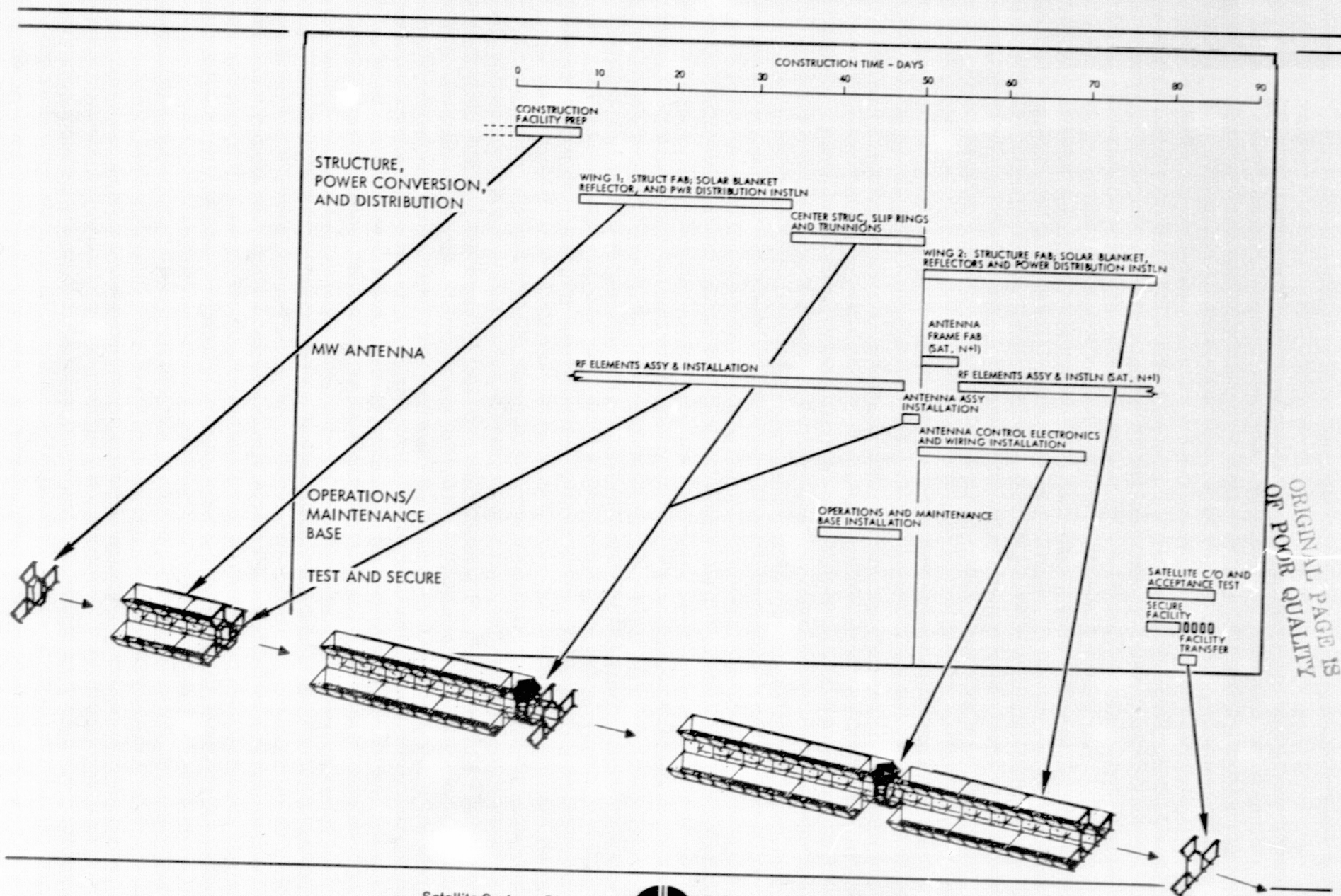
Each satellite wing consists of 12 bays 800-m long. These are constructed at the rate of one every two days using three 8-hour shifts per day. The structure and installation of the power conversion system of wing No. 1 is completed on the 34th day. While the wing No. 1 construction is taking place the MW antenna crews are proceeding with the assembly, test, and installation of the antenna elements into the antenna frame. The antenna assembly continues during the construction of the center section.

Subsequent to completion of wing No. 1 the construction facility constructs the longerons and frames in the center section, installs the slip rings, constructs the tension supports, installs the trunions, and installs power wiring in the center. Although 16 days are scheduled for this activity, the timeline requires only 12 days with two additional days scheduled for transfer of the antenna to the trunion mounts. Two days are allowed for contingencies.

Immediately upon completion of the center section primary structure the facilities for the operation and maintenance are installed and the first operational maintenance crew arrives to support installation of the antenna control electronics and satellite checkout, which takes place from day 50 through day 69.

Final satellite checkout and acceptance testing is completed on day 86. Use of the construction facility is completed on day 78 and flyaway transfer to the construction site of the next satellite occurs on day 84.

CONSTRUCTION SCHEDULE FOR ROCKWELL BASELINE SATELLITE, JUNE 1978



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ROCKWELL BASELINE SATELLITE CONSTRUCTION BASE(SCB),
JUNE 1978

Construction of the satellites takes place in GEO, each satellite being constructed at its designated longitudinal location. All construction activities are supported by a single integrated construction base which produces satellites at the rate of 4 per year (and later 5 per year) during the mature portion of the program. Upon completion of one satellite the base is moved to the operational location of the next satellite for construction of that satellite.

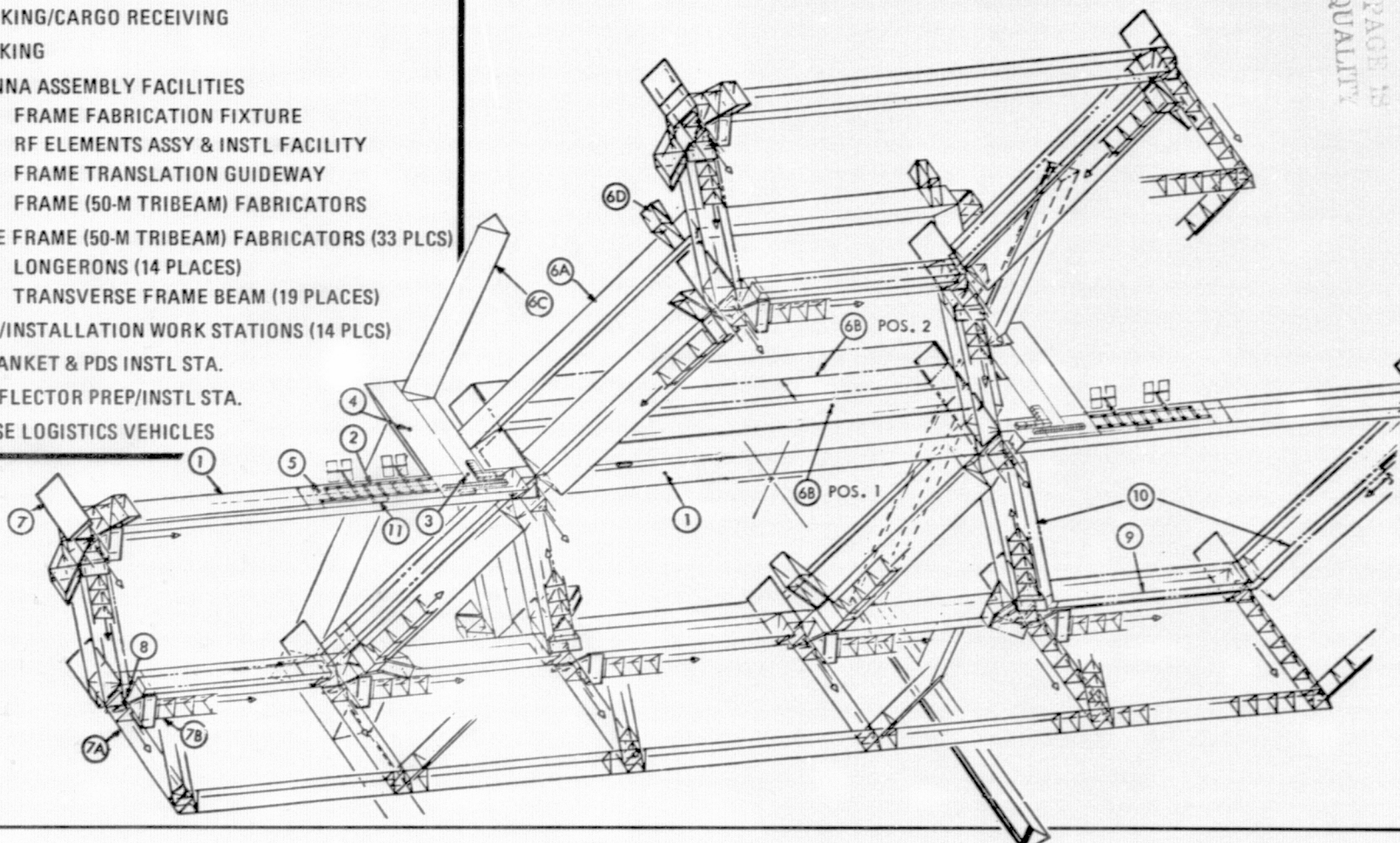
The construction base consists of the satellite construction fixture, the construction equipment, and the base support facilities and equipment. The construction fixture is a rugged heavy gage metal structure on which all elements of the construction base are mounted. The fixture constitutes the reference surfaces for the construction operations and the locating jig for the equipment which constructs/installs various elements of the satellite in situ.

The major construction equipment includes the 50 m tribeam fabricators; the deployment equipment for the solar cell blankets, the solar reflector panels, the power distribution conductors, the cables for retention of the solar blankets, and the structure tensioning cables; the assembly facility for the MW antenna mechanical modules; and the equipment for installation of the MW antenna elements into the antenna frame. The location of most of these elements is identified on the chart.

ROCKWELL BASELINE SATELLITE CONSTRUCTION BASE (SCB), JUNE 1978

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- ① CONSTRUCTION FIXTURE
- ② BASE SUPPORT FACILITIES & EQUIP (2 EA)
340-CREW MEMBER SUPPORT
BASE SUBSYSTEM, MAINTENANCE SHOPS
BASE MGMT, COMM., CONTROL, LOGISTICS
- ③ WAREHOUSE
- ④ EOTV DOCKING/CARGO RECEIVING
- ⑤ POTV DOCKING
- ⑥ MW ANTENNA ASSEMBLY FACILITIES
 - ⑥A FRAME FABRICATION FIXTURE
 - ⑥B RF ELEMENTS ASSY & INSTL FACILITY
 - ⑥C FRAME TRANSLATION GUIDEWAY
 - ⑥D FRAME (50-M TRIBEAM) FABRICATORS
- ⑦ SATELLITE FRAME (50-M TRIBEAM) FABRICATORS (33 PLCS)
 - ⑦A LONGERONS (14 PLACES)
 - ⑦B TRANSVERSE FRAME BEAM (19 PLACES)
- ⑧ BEAM FAB/INSTALLATION WORK STATIONS (14 PLCS)
- ⑨ SOLAR BLANKET & PDS INSTL STA.
- ⑩ SOLAR REFLECTOR PREP/INSTL STA.
- ⑪ INTRA-BASE LOGISTICS VEHICLES



ROCKWELL BASELINE SCB MASS SUMMARY, JUNE 1978

BEAM MACHINES: 234 x 700 KG	163800 KG
TRIBEAM FABRICATORS: 39 x 2100 KG	81900
REFLECTOR INST. EQUIP: 6 x 20,000 KG	120000
SOLAR BLANKET DISPENSER: 73 x 2000 KG	146000
CABLE & CATENARY DISPENSER: 307 x 200 KG	61400
CABLE & CATENARY ATTACH. MACH: 79 x 500 KG	39500
	<u>612600</u>
MICROWAVE ANTENNA	450000
FABRICATION FIXTURE	2000000
BOOM MOUNTED MANIP. MODULES: 36 x 5000	
BEAM STATION LOG. VEHICLE: 6 x 5000	
CREW LOGISTICS VEHICLE 16 x 5000	
58 x 5000	<u>290000</u>
	3352600
HABITAT	2040000
BASE POWER SUPPLY	<u>1000000</u>
TOTAL	6392600 KG

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CURRENT CONSTRUCTION GUIDELINES

JUNE '78

- ✓ ● SINGLE INTEGRATED SCB
- ✓ ● CONTINUOUS STRUCTURE FABRICATION
- ✓ ● SOLAR CONVERTER INSTALLATION SIMULTANEOUS WITH
STRUCTURAL FAB
- ANTENNA ASSEMBLED IN PLACE CONCURRENT WITH
CONSTRUCTION OF SECOND WING
- ✓ ● NO SCHEDULED EVA
- 180 DAY CONSTRUCTION SCHEDULE



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183

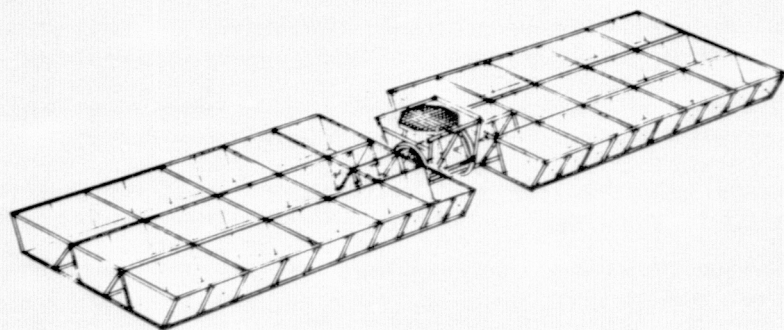
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COPLANAR SATELLITE AND CONSTRUCTION OPTIONS

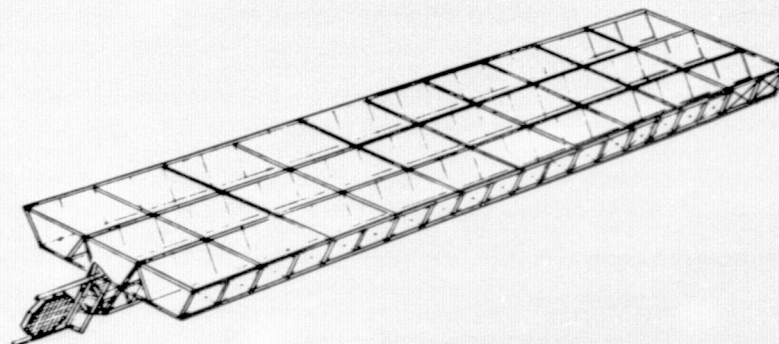
This chart shows the 3-trough coplanar satellite with center mounted antenna and with an end mounted antenna together with their applicable optional construction techniques. (The effect of increasing the satellite width to four troughs, with attendant reduction in lengths, is shown on the next two charts.) The serpentine construction technique offers the potential of a small SCB and small crew size but offers little flexibility for reducing construction time since each trough is constructed serially; also it is less appropriate for the center mounted antenna configuration. The single pass construction technique has somewhat the opposite characteristics.

SATELLITE AND CONSTRUCTION OPTIONS

BASIC SATELLITE CONFIGURATIONS

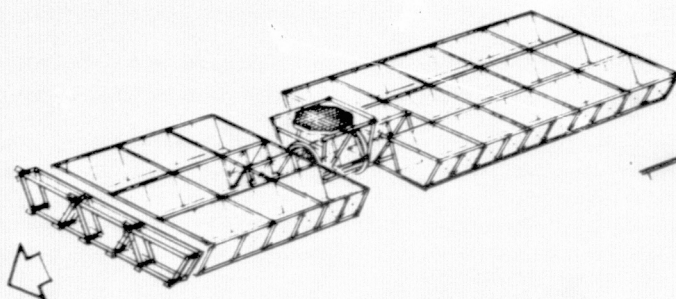


CENTER MOUNTED ANTENNA

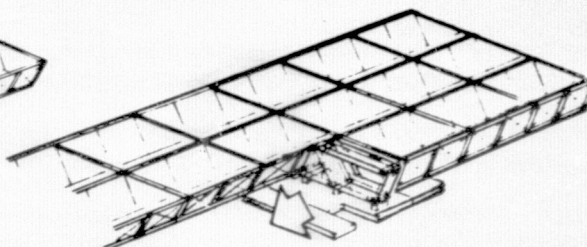


END MOUNTED ANTENNA

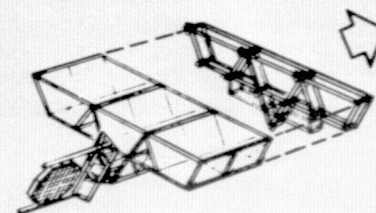
CONSTRUCTION CONCEPTS



SINGLE PASS



MULTI-PASS SERPENTINE



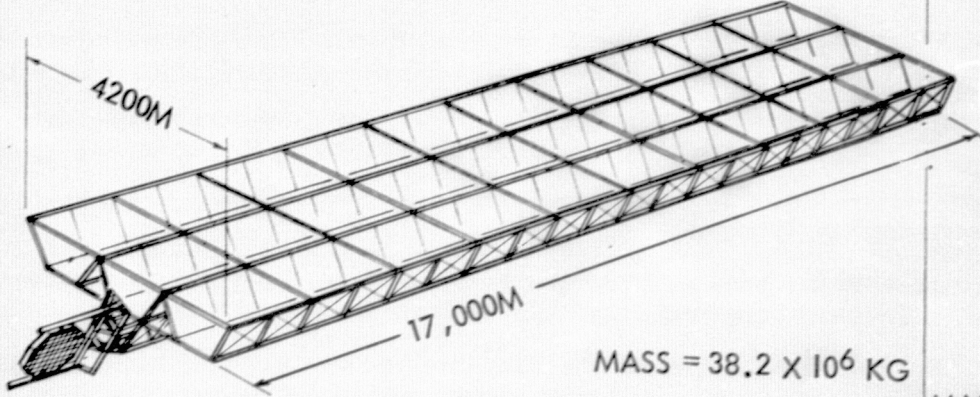
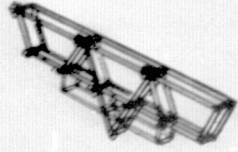
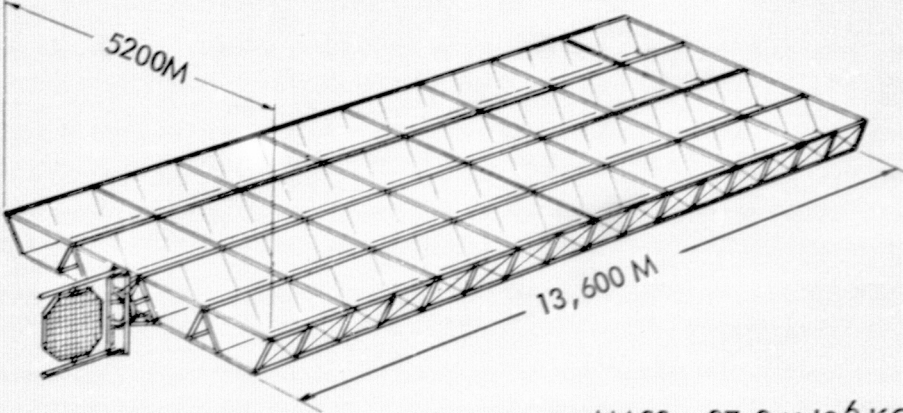
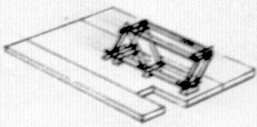
SINGLE PASS

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3- AND 4- TROUGH SATELLITES WITH END MOUNTED ANTENNA

Three and four trough versions of the satellite end-mounted antenna configurations are shown. The solar blanket area is the same for each version. Two different construction concepts have been identified; parallel build single pass for the three troughs and serpentine for the four troughs. As is shown in later charts, the single pass concept possess potential for shortening the nominal 180 day construction schedule; the serpentine concept, because of sequential trough construction, requires the entire 180 days. While the mass of the SCB used for serpentine construction is slightly less than for the single pass SCB, the serpentine SCB, featuring a large platform with sliding sections, is more complex.

3- AND 4-TROUGH SATELLITES WITH END MOUNTED ANTENNA

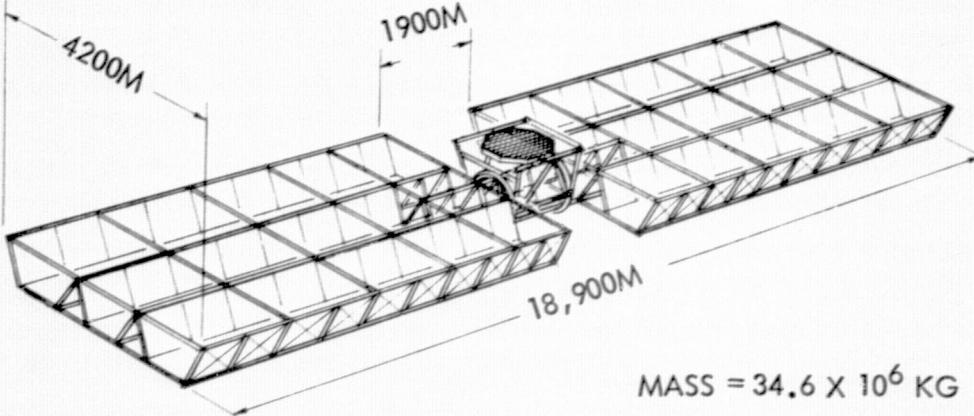
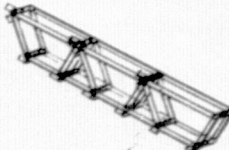
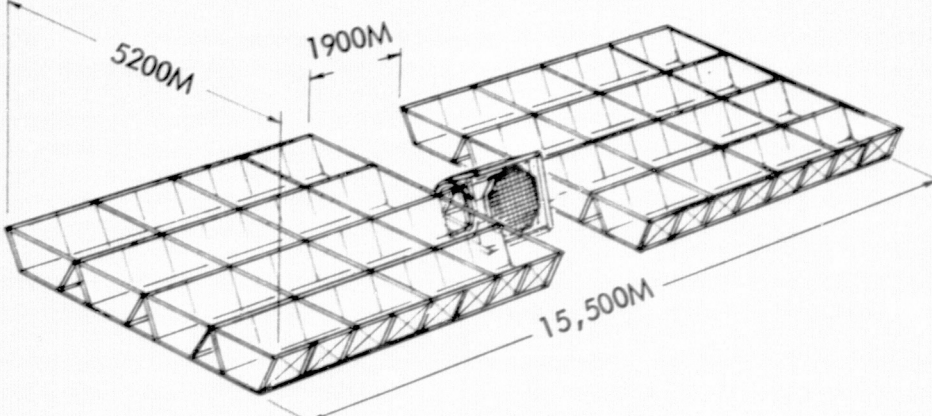
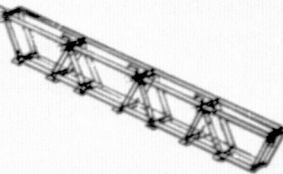
SATELLITE CONFIGURATION	SCB CONFIGURATION	CONSTR CONCEPT
 4200M 17,000M MASS = 38.2×10^6 KG	 MASS = 5.3×10^6 KG	PARALLEL BUILD SINGLE PASS
 5200M 13,600M MASS = 37.2×10^6 KG	 MASS = 5.2×10^6 KG	SERPENTINE 4 PASSES

3- AND 4- TROUGH SATELLITES WITH CENTER MOUNTED ANTENNA

The center mounted antenna is shown for the same trough configurations displayed on the previous chart. The difference in satellite mass for these configurations as compared to the end-mounted versions is largely attributable to power distribution. (Parallel build, single pass construction was selected for these configurations since the complexity associated with serpentine build of a center-mounted antenna configuration appeared to be excessive.)

From a constructability standpoint the 3-trough satellite is more desirable than the 4-trough configuration because the SCB is narrower, of lower mass, and requires a smaller crew size.

3- AND 4-TROUGH SATELLITES WITH CENTER MOUNTED ANTENNA

SATELLITE CONFIGURATION	SCB CONFIGURATION	CONSTR CONCEPT
 MASS = 34.6 X 10⁶ KG	 MASS = 5.3 X 10⁶ KG	SINGLE PASS
 MASS = 35.8 X 10⁶ KG	 MASS = 6.5 X 10⁶ KG	SINGLE PASS

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SCENARIO FOR
SERPENTINE CONSTRUCTION OF A
COPLANAR SATELLITE



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191

190
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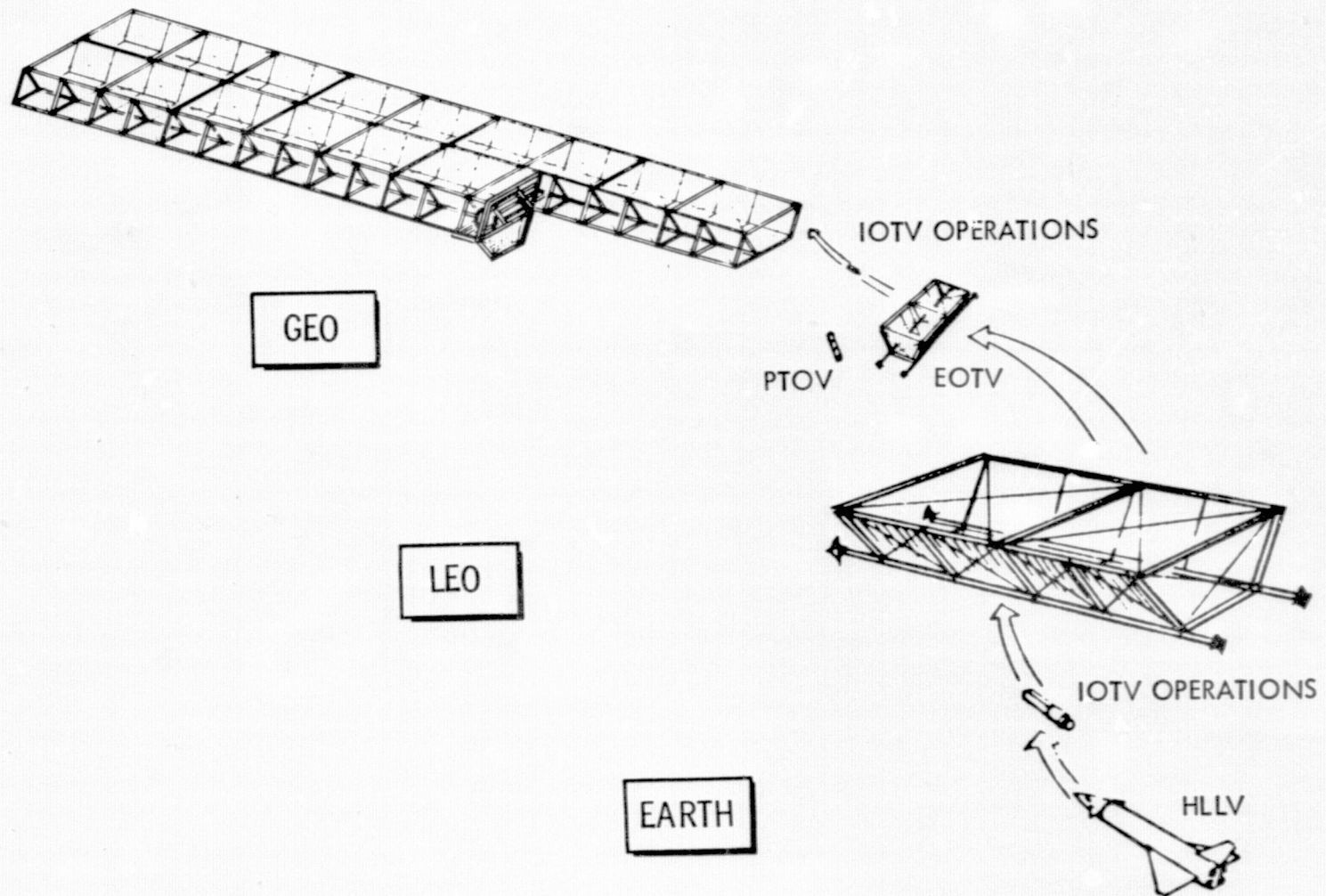
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CONSTRUCTION OVERVIEW

This scenario overview shows the primary operations required to support satellite construction. Initially, construction material is delivered to LEO by HLLV's and transferred directly to EOTV's by the IOTV's. The EOTV's transit to GEO with their cargo, which is similarly transferred to the satellite construction base. The construction crew and their chemically propelled POTV's arrive in LEO via HLLV's, where the stages are mated. The POTV's then proceed to GEO with their crew and are utilized for subsequent crew return to LEO.

"Down" cargo is transferred to LEO by EOTV's. This cargo and POTV's returning from GEO are transported to earth by HLLV's.

CONSTRUCTION OVERVIEW

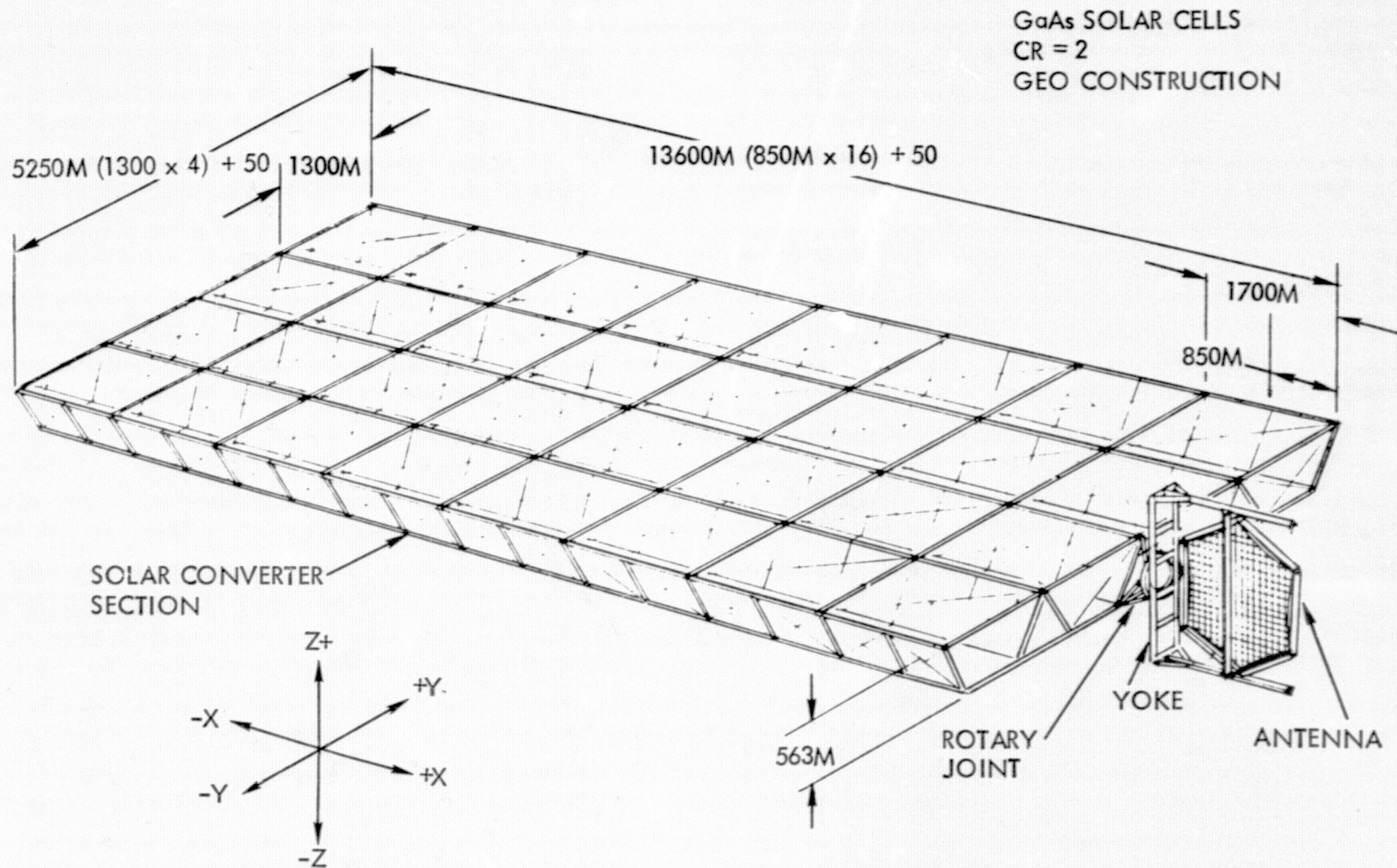


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REPRESENTATIVE COPLANAR SATELLITE CONCEPT

The satellite configuration utilized to describe the construction concept consists of four parallel solar converter troughs, each trough having 16 structural bays as shown. The bottom of the bays are 700 m wide and have solar cell blankets installed on the underneath side of the lower cross beams. Reflectors are installed on the inside of the diagonal structural beams to provide a CR of 2. The primary structural members are 50 m tribeams constructed from 2 m cap and cross beams utilizing tribeam fabricators. The tribeam fabricators are capable of producing any length of 50 m beam as a continuous member.

REPRESENTATIVE COPLANAR SATELLITE

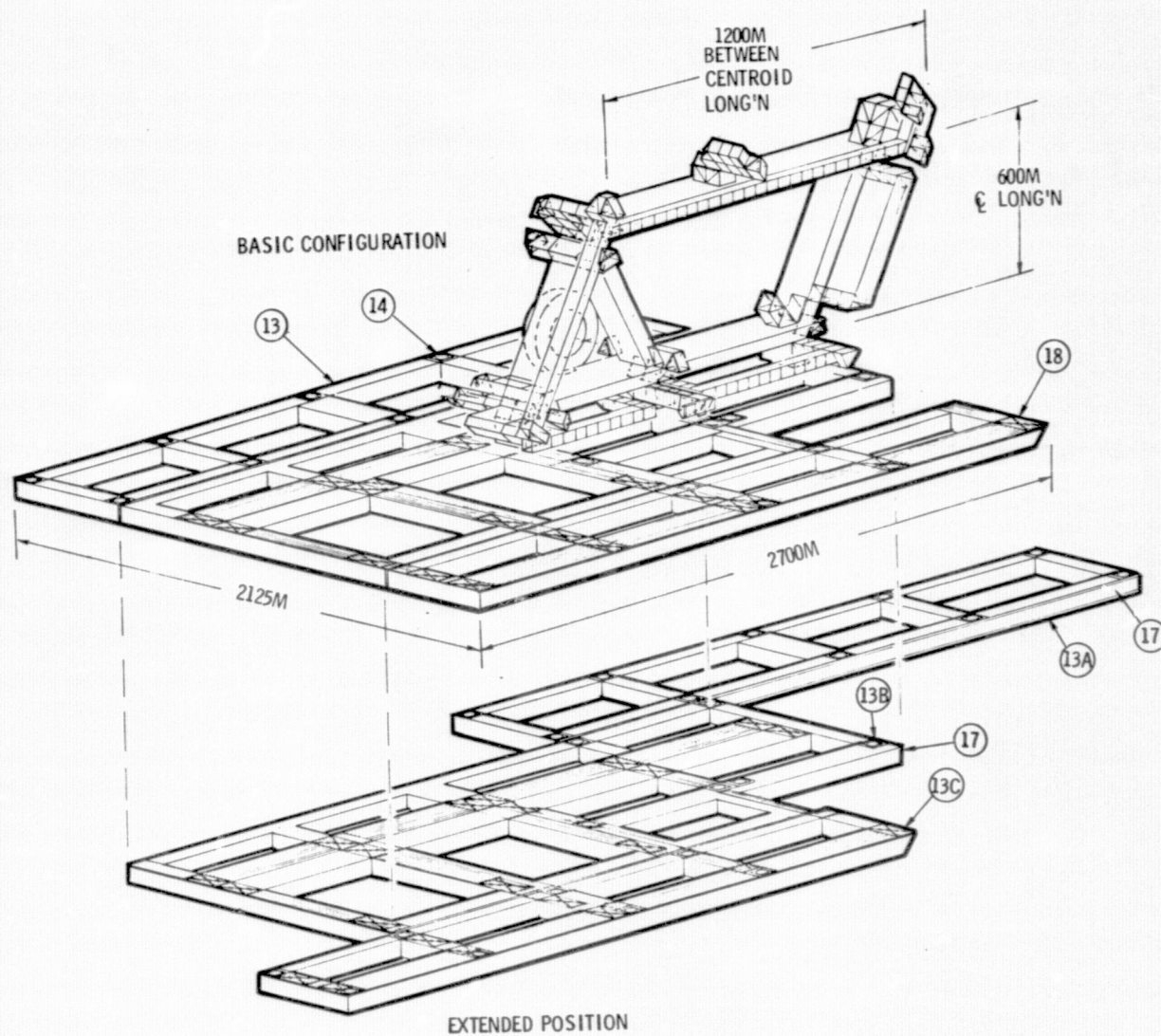


SATELLITE CONSTRUCTION BASE (SCB)
(SERPENTINE CONSTRUCTION)

The translation platform is shown here with the construction facility attached to the tracks, (18) by the three translating carriages. The platform consists of three sections attached to one another by means of sliding guideways, (17) which permit lateral relative movement during the repositioning operations as shown on the bottom figure and as described in other charts. The elevating frame attach fittings, (14), are used to secure the platform to the partially completed satellite and to thus permit movement of the construction facility relative to the satellite.

(The callouts indicated by the circled numbers are identified on the following chart)

SATELLITE CONSTRUCTION BASE (SCB) (SERPENTINE CONSTRUCTION)



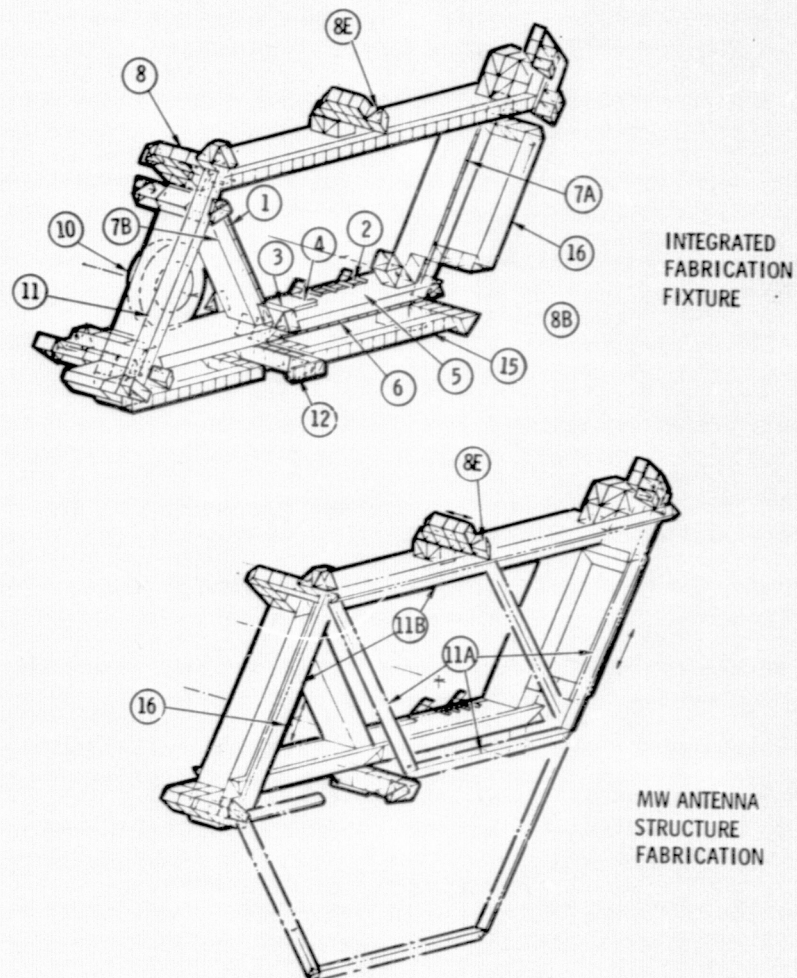
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INTEGRATED FABRICATION FIXTURE

The fixture shown is for the serpentine construction concept. This entails beam fabrication from each side of the fixture as shown, for example by the duplicate sets of longeron beam fabricators, denoted by (8A) and (8B). In addition to fabricators for the basic satellite structure, fabricators for the antenna frame are shown (8E). The concept for building the antenna frame a half at a time will be sequently covered. Antenna RF elements are installed from the two assembly and installation stations, (15). Crew, power, warehousing and receiving facilities are grouped on the central portion of the fixture, indicated by (2) through (5). The rotating joint assembly fixture, (10), is located in the triangle formed by the two left diagonals.

The fixture is mounted on a platform in tracks which provide for transverse and longitudinal movement. The movement is effected by means of 3 translation carriages, (12), which are attached to tracks in the platform, shown in the last chart.

INTEGRATED FABRICATION FACILITY (SERPENTINE CONSTRUCTION)



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- ① CONSTRUCTION FIXTURE
- ② MAIN SUPPORT FACILITIES & EQUIPMENT
- ③ WAREHOUSE
- ④ CARGO RECEIVING
- ⑤ POTV DOCKING
- ⑥ SOLAR BLANKET & PDS INST STATION
- ⑦A 2 REFLECTOR INSTALLATION STATIONS ON UNDERSIDE (+X) OF DIAGONAL
- ⑦B 2 REFLECTOR INSTALLATION STATIONS, ONE EACH ON +Y & -Y SIDE OF DIAGONAL BRACE
- ⑧ BEAM FAB/INST WORK STATIONS
- ⑧A LONGERON TRI-BEAM FABRICATORS, +Y (3 PLACES (5 PLACES FOR TROUGH 1))
- ⑧B LONGERON TRI-BEAM FABRICATORS, -Y (3 PLACES (5 PLACES FOR TROUGH 1))
- ⑧C DIAGONAL TRI-BEAM FABRICATORS, (2 PLACES (3 PLACES FOR TROUGH 1))
- ⑧D TRANSVERSE BEAM FABRICATORS (3 PLACES)
- ⑧E ANTENNA FRAME TRI-BEAM FABRICATORS (6 PLACES)
- ⑨ INTRABASE LOGISTICS VEHICLES

- ⑩ ROTATING JOINT ASSEMBLY FIXTURE
- ⑪ ANTENNA FRAME FAB FIXTURE
- ⑪A LOWER HALF
- ⑪B UPPER HALF
- ⑫ CONSTRUCTION FIXTURE TRANSLATION CARRIAGE (3 PLACES)
- ⑬ TRANSLATION PLATFORM
- ⑬A SECTION A
- ⑬B SECTION B
- ⑬C SECTION C
- ⑭ ELEVATING FRAME ATTACH FITTING (18 PLACES)
- ⑮ RF ASSY & INST STATION (2 PLACES)
- ⑯ ANTENNA TRANSLATION GUIDEWAYS (2 PLACES)
- ⑰ PLATFORM SLIDING GUIDEWAYS (2 PLACES)
- ⑱ FACILITY TRAVELWAY
- ⑲ INTEGRATED FABRICATION FACILITY



SCB MASS SUMMARY
(Serpentine Concept)

This chart is self-explanatory.

SCB MASS SUMMARY (SERPENTINE CONCEPT)

TRIBEAM FABRICATORS	141120 KG
REFLECTOR INST. EQUIPMENT	80000
SOLAR BLANKET DISPENSORS (INCL. CABLE & CATENARIES)	115200

FIXTURE STRUCTURE	819000
TRANSLATING PLATFORM	1412000
RF ASSY/INSTLN. STATIONS	160000
LOG. VEHICLES AND MANNED MANIPULATORS	198000

HABITAT AND POWER SUPPLY	<u>1222000</u>
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TOTAL	4152320 KG
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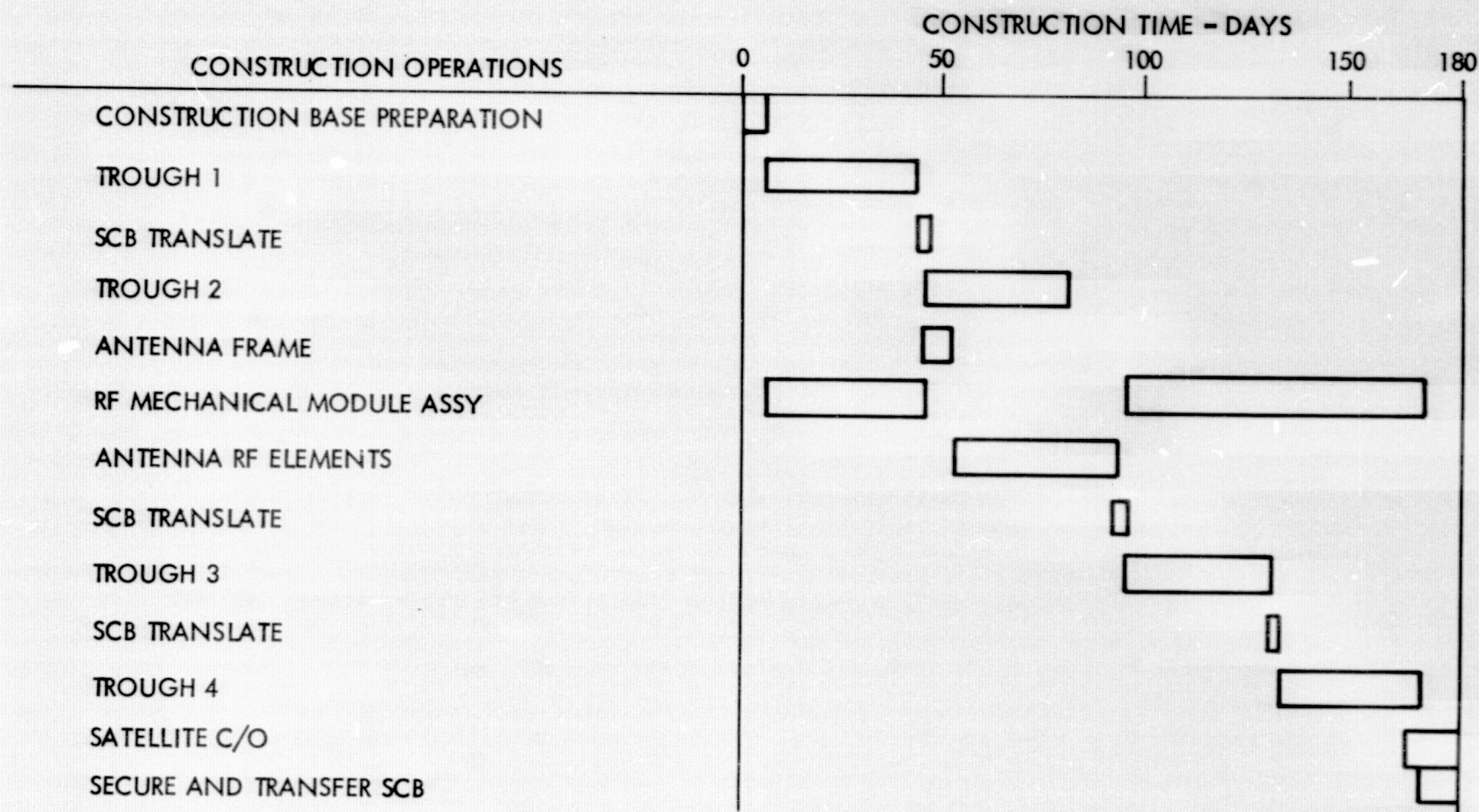
TOTAL (INC. 25% GROWTH)	5190000 KG
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Nth SATELLITE SERPENTINE CONSTRUCTION SCHEDULE
(END MOUNTED TENSION WEB ANTENNA)

The serpentine construction concept requires that the troughs be constructed in sequence as shown. This essentially becomes the pacing activity. The antenna frame is fabricated on the fixture during the second pass as previously described, followed by installation of the antenna RF elements. Assembly of RF mechanical modules occurs during most of the 180 days except when RF elements are being installed, since these two operations are conducted by the same crew. Upon completion of the antenna, the assembly of mechanical modules for the next satellite commences, as shown by the second bar. This provides a greater time span for this task with a resultant reduction in crew and equipment.

Nth SATELLITE SERPENTINE CONSTRUCTION SCHEDULE (END-MOUNTED TENSION WEB ANTENNA)

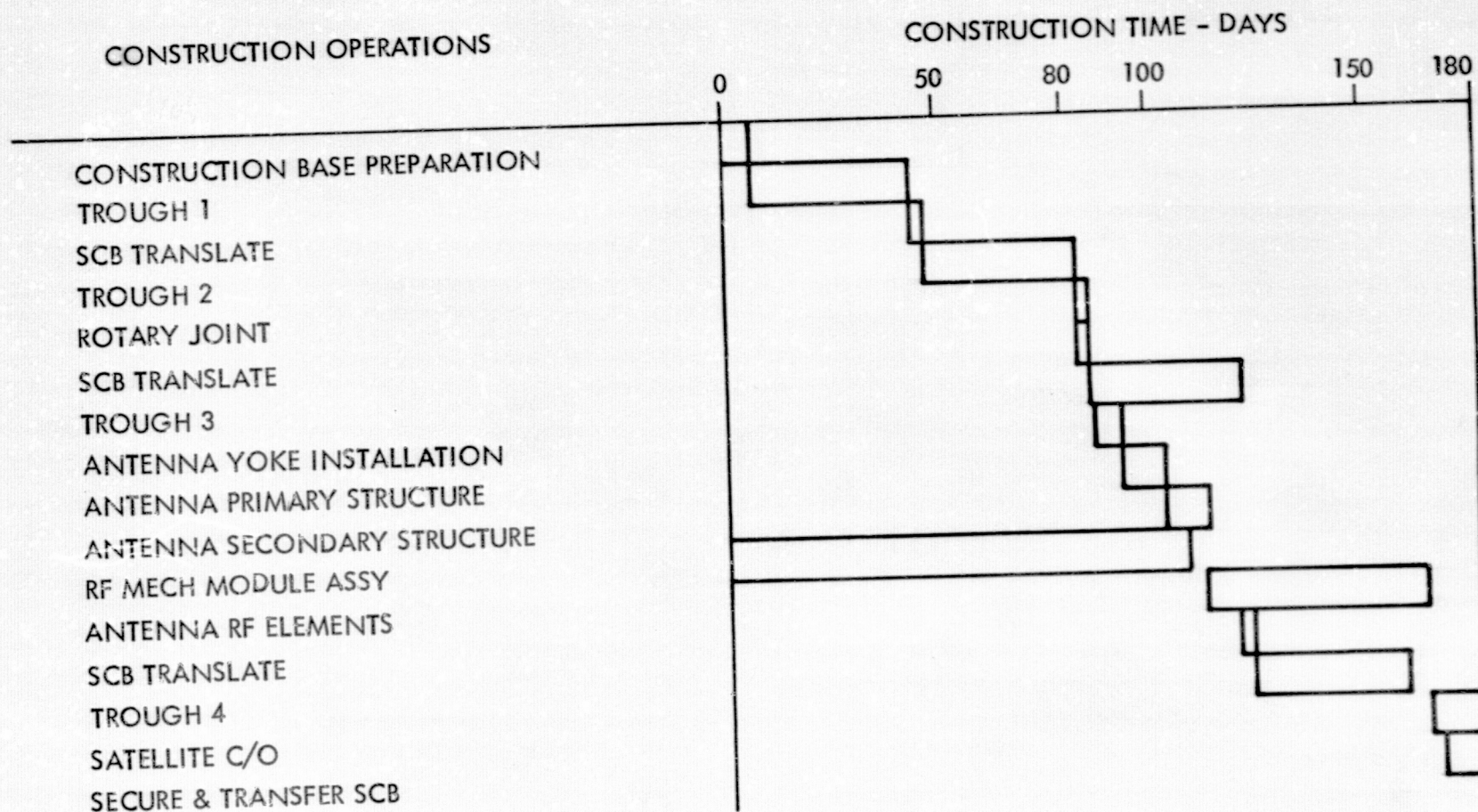


Nth SATELLITE (SERPENTINE CONSTRUCTION) SCHEDULE

This timeline differs from the one previously shown for the 4-trough configuration, primarily in antenna construction, where the space frame antenna replaces the tension web version. Additional time must be provided for antenna construction, since the antenna is composed of a beam type primary structure, fabricated by gantry-mounted beam fabricators. This structure must then be faced with a secondary structure upon which the RF elements will be mounted. This additional step delays the antenna RF installation task and results in the overall antenna construction being the pacing item.

However, since antenna RF installation is conducted in parallel with construction of the fourth trough, the overall schedule of 180 days is not impacted. The timeline also requires relocation of the time allocated to mechanical module assembly. Although the total allocated time is the same as in the previous timeline, the activity is conducted prior to construction of the antenna and consists of assembling modules for the Nth satellite only.

Nth SATELLITE SERPENTINE CONSTRUCTION SCHEDULE (END MOUNTED SPACE FRAME ANTENNA)



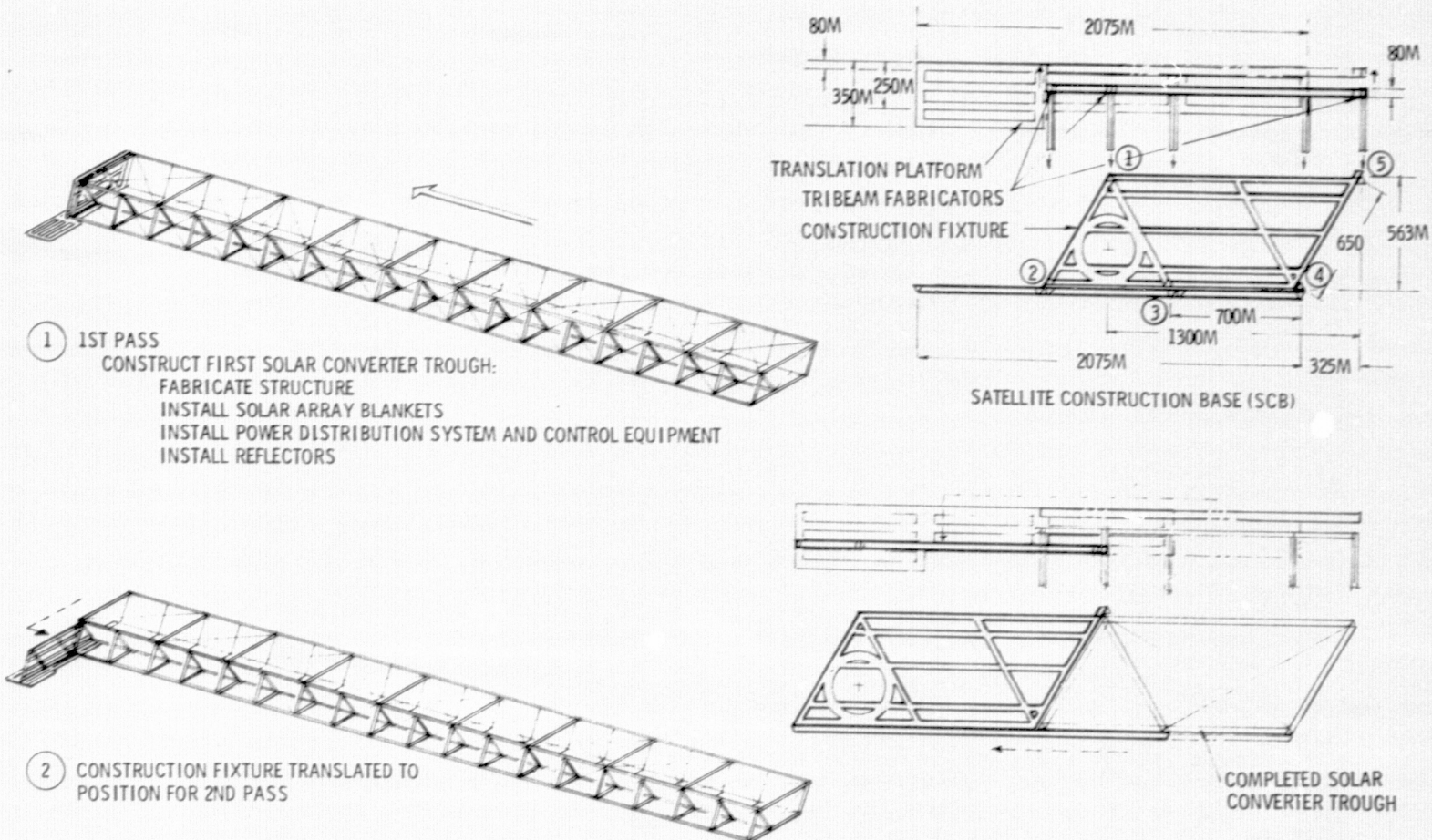
TROUGH 1 CONSTRUCTION

Five tribeam fabricators for making longerons are indicated in the upper right hand view on the chart. All five are used in the first pass as trough 1 is built. In constructing the remaining troughs only three fabricators (1, 2, 3) are required. The longerons are continuous members for the entire length of the satellite. Only four tribeam fabricators are required for making the cross beams except for the first pass which requires six. The cross beams are continuous for their respective lengths. In making the solar converter troughs the initial frame is constructed using the fixture as the tooling jig. The longerons are then fabricated away from the face of the SCB moving the frame with them. As they move out, the solar blankets, reflectors, power feeders and other elements of the solar converter are dispensed. At the completion of the length of each bay, fabrication in the longitudinal direction is stopped, the beams for the next frame (which have been fabricated during construction of the longerons) are connected in place to the longerons and the solar blankets and reflectors are tensioned between the frames to complete the bay.

The translating platform supports the fixture assembly and all SCB facilities. It is designed to move with the rest of the SCB in the longitudinal direction but can also be locked to the completed structure to perform its functions at the end of each trough. The fixture assembly is mounted in tracks on the platform which permit movement of the fixture both laterally and longitudinally. The platform extends 775 m wider than the structure of the trough being constructed. After the last frame of trough 1 is completed the translation platform is clamped to the longerons of trough 1. The fixture assembly is first translated in the longitudinal direction to clear the last frame, then translated laterally to be in line for construction of trough 2. It is then translated longitudinally back along the side of trough 1 to be in position for fabrication of the first frame of trough 2. After the first frame of trough 2 is completed and tied into the last frame of trough 1, the translation platform is released from the trough 1 longerons and translated to the left to again be in the position with respect to the fixture assembly shown on the right hand side of the chart.



TROUGH 1 CONSTRUCTION



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TROUGH 2 AND MW ANTENNA FRAME CONSTRUCTION

This series of charts illustrate the serpentine coplanar satellite configuration and construction concept. The construction fixture is designed for simultaneous fabrication of the antenna on the -Y face while the second trough is being fabricated from the +Y face. The SCB geometry and the sequence of construction of the solar converter portion of the satellite make it necessary that the antenna and rotary joint installations be completed prior to starting construction of the third trough. The rotary joint can be assembled and checked out independently on the special circular jig any time during construction of the first two troughs. Antenna frame fabrication is initiated simultaneously with start of construction of trough No. 2. Installation of the RF elements is started as soon as the tension web installation is completed. Fabrication of the antenna yoke and rotary joint standoff support structure occurs after completion of trough No. 2 since those operations take place on the +Y side of the construction fixture.

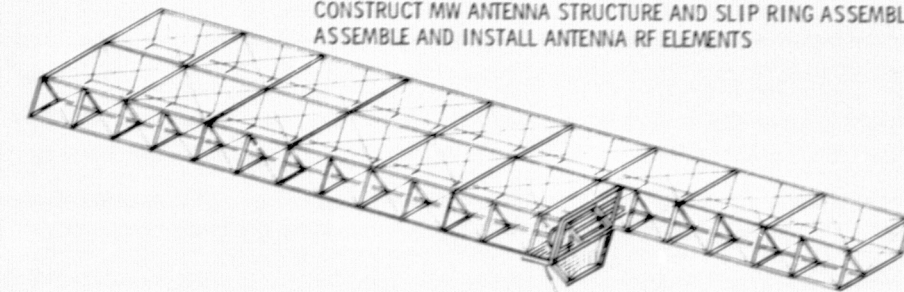
The 50-m tribeam fabricators produce the structure for the antenna frame. Referring to the middle view on the chart, the lower one-half portion of the frame is fabricated first using the part of the fixture indicated by the heavy dashed lines. This half-frame is closed out across the top with the temporary 50-m tribeam tie bar indicated for stabilization during the remainder of the assembly operations. This frame half is then translated to a position below the translation platform (arrows and phantom outline in the middle view) and the upper half is fabricated using the heavy solid line part of the fixture, and the two-halves are joined to form the hex-shaped frame.

The lower right-hand view shows that the SCB provides for up-down translation of the antenna in the X-Z plane. The position of the tie bar is variable from the X-Y plane of the base of the translating platform to the X-Y plane of the upper cross frame of the construction fixture. This permits installation of the tension web, installation of the RF mechanical modules (antenna upper-half installed from the upper RF assembly and installation facility, antenna bottom half installed from the lower RF facility), and alignment of the completed antenna with the rotary joint. Assembly and checkout of the subarrays and mechanical modules is also accomplished at the RF facilities.

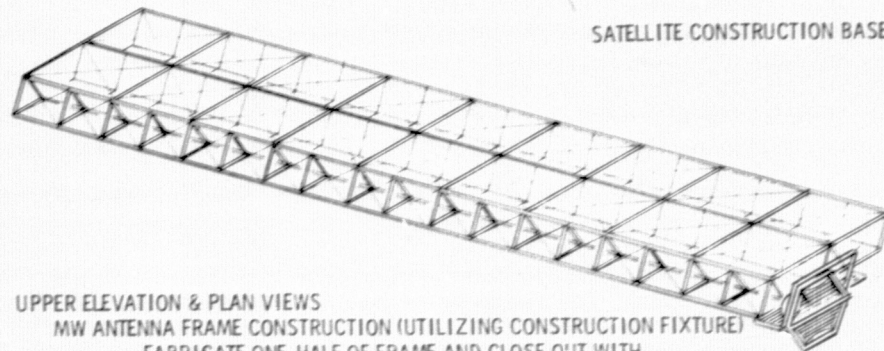
TROUGH 2 AND MW ANTENNA CONSTRUCTION & RF INSTALLATION

③ & ④ 2ND PASS

CONSTRUCT SECOND SOLAR CONVERTER TROUGH
CONSTRUCT MW ANTENNA STRUCTURE AND SLIP RING ASSEMBLY
ASSEMBLE AND INSTALL ANTENNA RF ELEMENTS



SATELLITE CONSTRUCTION BASE

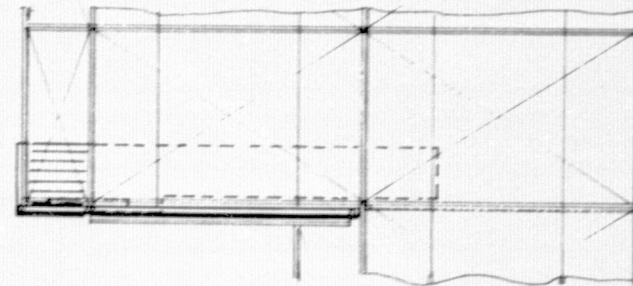


UPPER ELEVATION & PLAN VIEWS

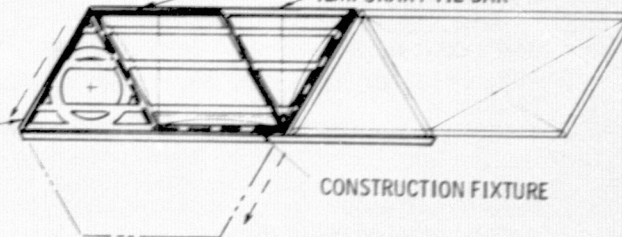
MW ANTENNA FRAME CONSTRUCTION (UTILIZING CONSTRUCTION FIXTURE)
FABRICATE ONE-HALF OF FRAME AND CLOSE OUT WITH
TEMPORARY TIEBAR (80M GIRDER)
TRANSLATE HALF FRAME TO LOWER POSITION
CONSTRUCT UPPER HALF OF FRAME AND JOIN TO LOWER HALF
INSTALL TENSION WEB

LOWER ELEVATION VIEW

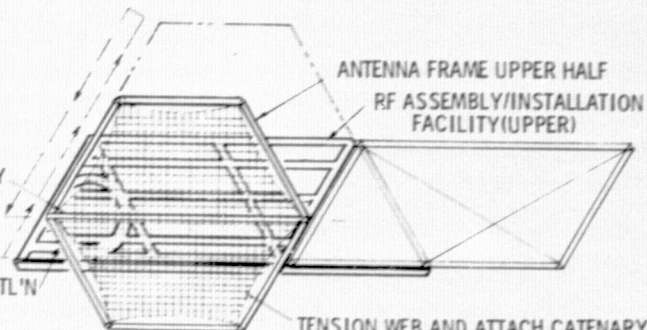
ANTENNA RF ELEMENTS ASSEMBLY AND INSTALLATION (RF ASSEMBLY/
INSTALLATION FACILITY LOCATED ON SPS CONSTRUCTION BASE)
ASSEMBLE AND C/O SUBARRAY
ASSEMBLE AND C/O MECHANICAL MODULES AND PWR DISTR ELEMENTS
INSTALL MECHANICAL MODULES AND POWER DISTRIBUTION
SYSTEM ON ANTENNA TENSION WEB



ANTENNA FRAME LOWER HALF
TEMPORARY TIE BAR



CONSTRUCTION FIXTURE



TEMPORARY
TIE BAR

RF ASSY/INSTLN
FAC (LOWER)

ANTENNA FRAME UPPER HALF
RF ASSEMBLY/INSTALLATION
FACILITY(UPPER)

TENSION WEB AND ATTACH CATENARY

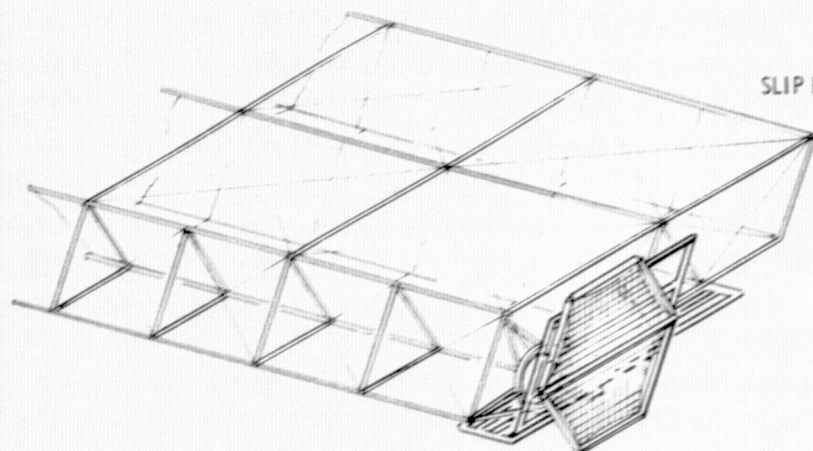
SHEET 2 OF 6

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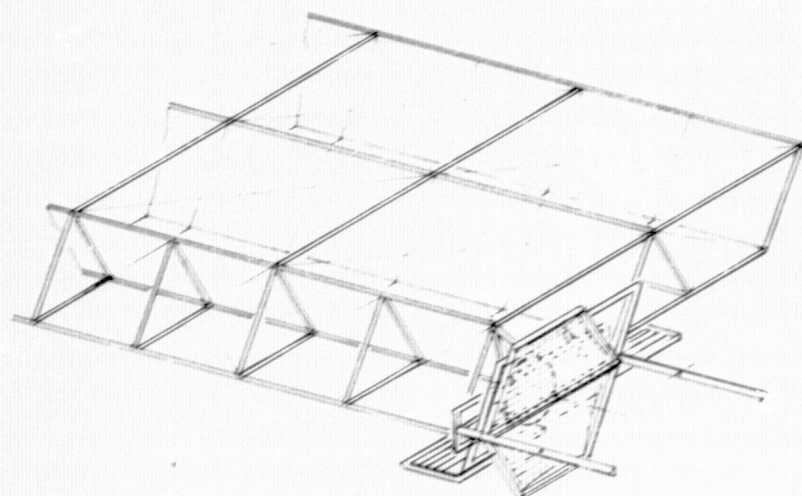
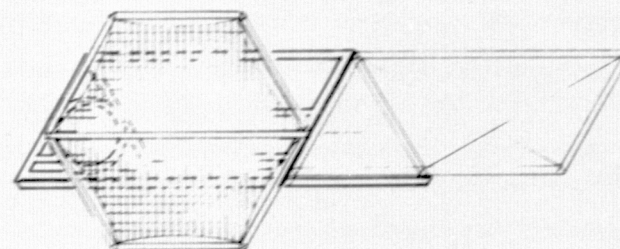
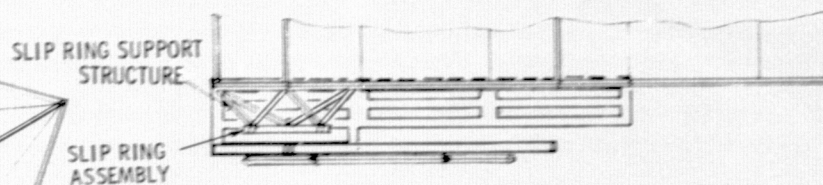
ANTENNA ROTARY JOINT AND YOKE CONSTRUCTION

Upon completion of trough No. 2 the slip ring supports and antenna yoke shown on the chart are fabricated using one of the available 50-m tribeam fabricators. The slip ring supports are joined to the solar converter structure and the slip-ring is removed from its jig and installed on the supports. The base of the antenna yoke is fabricated parallel to the translating base, the antenna is translated to be centered in the projection of the base, and the trunnion support arms are built out in the -Y direction to pass through extensions of the antenna center line on which the trunnions are located. Guideways are provided along the trunnion support arms which engage the trunnions as the support arms are fabricated outward. The translation platform is used to move the center line of the antenna/yoke coincident with the center line of the rotary joint and the yoke is attached to the rotary joint.

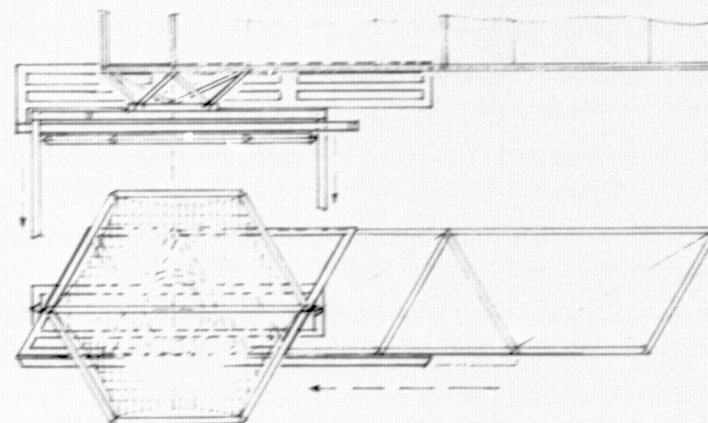
ANTENNA ROTARY JOINT & YOKE CONSTRUCTION



5 SLIPRING SUPPORT STRUCTURE INSTALLED



6 ANTENNA CENTER TRANSLATED TO BE COINCIDENT WITH SLIPRING CENTER VIA TRANSLATION PLATFORM
TRUNNION SUPPORT BASE PLATFORM FABRICATED ONTO OUTER SLIPRING



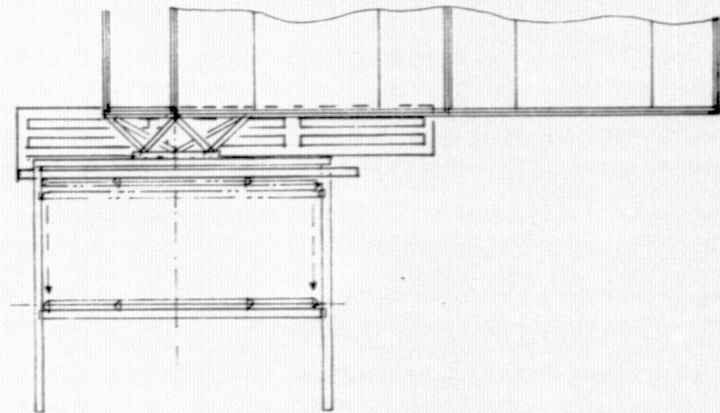
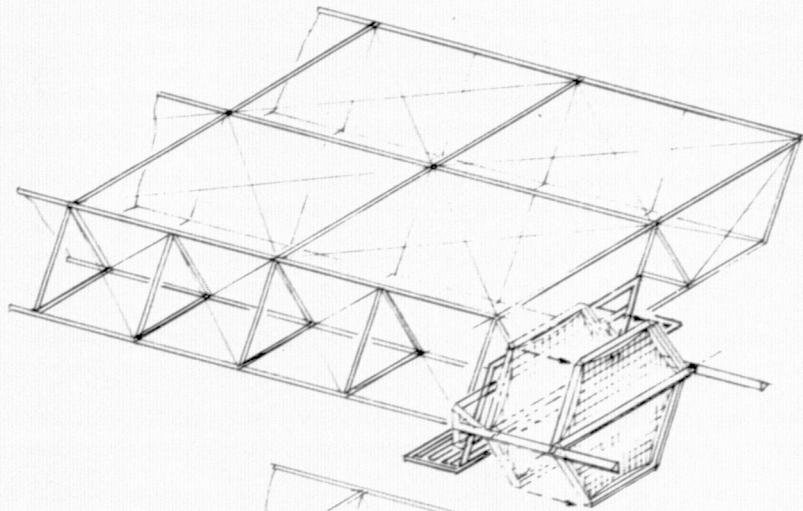
TRUNNION SUPPORT ARMS BUILT OUT FROM BASE PLATFORM
BRACING STRUCT. FOR TSA ADDED
CRAWLER SYSTEM INSTALLED ON TEMPORARY ANTENNA FRAME TIEBAR

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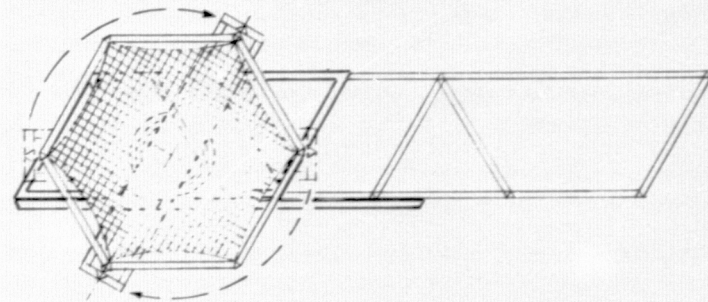
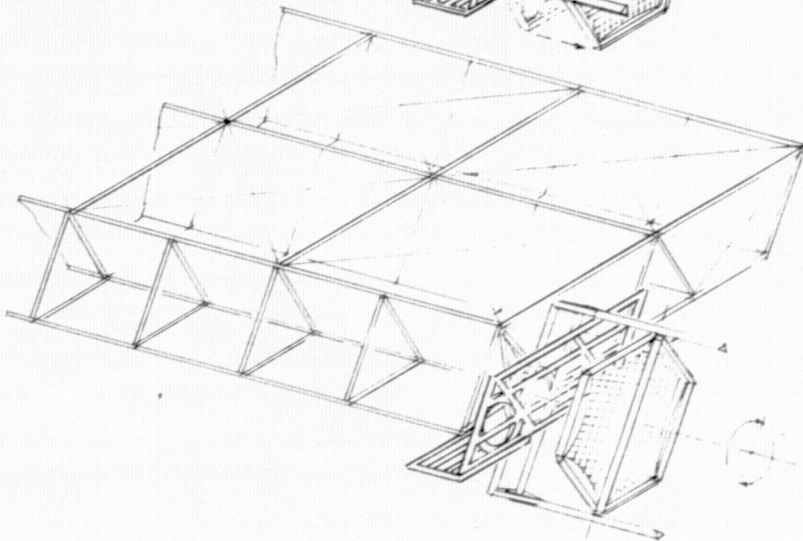
ANTENNA MOUNTING AND ROTATION

Referring to the chart a crawler system installed on the temporary tie bar is used to translate the antenna along the trunnion guideways to its gimble plane. The gimbles are secured, azimuth control elements are installed, and electrical power, information management and control system connections are made utilizing the crawler system. Since the slip ring and yoke base are necessarily constructed on opposite sides of the construction fixture it is necessary to rotate the slip ring and antenna assembly 120° as shown to release the construction base. The construction base is then translated into position to start construction of trough No. 3 (see next chart) through a series of operations similar to those described for moving from trough No. 1.

ANTENNA MOUNTING & ROTATION



- 7 ANTENNA TRANSLATED TO GIMBAL PLANE VIA
CRAWLER SYSTEM ON TIE GIRDER
GIMBAL PICKUP ASSY ADDED
ANTENNA SECURED TO TRUNNION SUPPORT ARMS



- 8 ANTENNA/TRUNNION ARMS ROTATED 120°

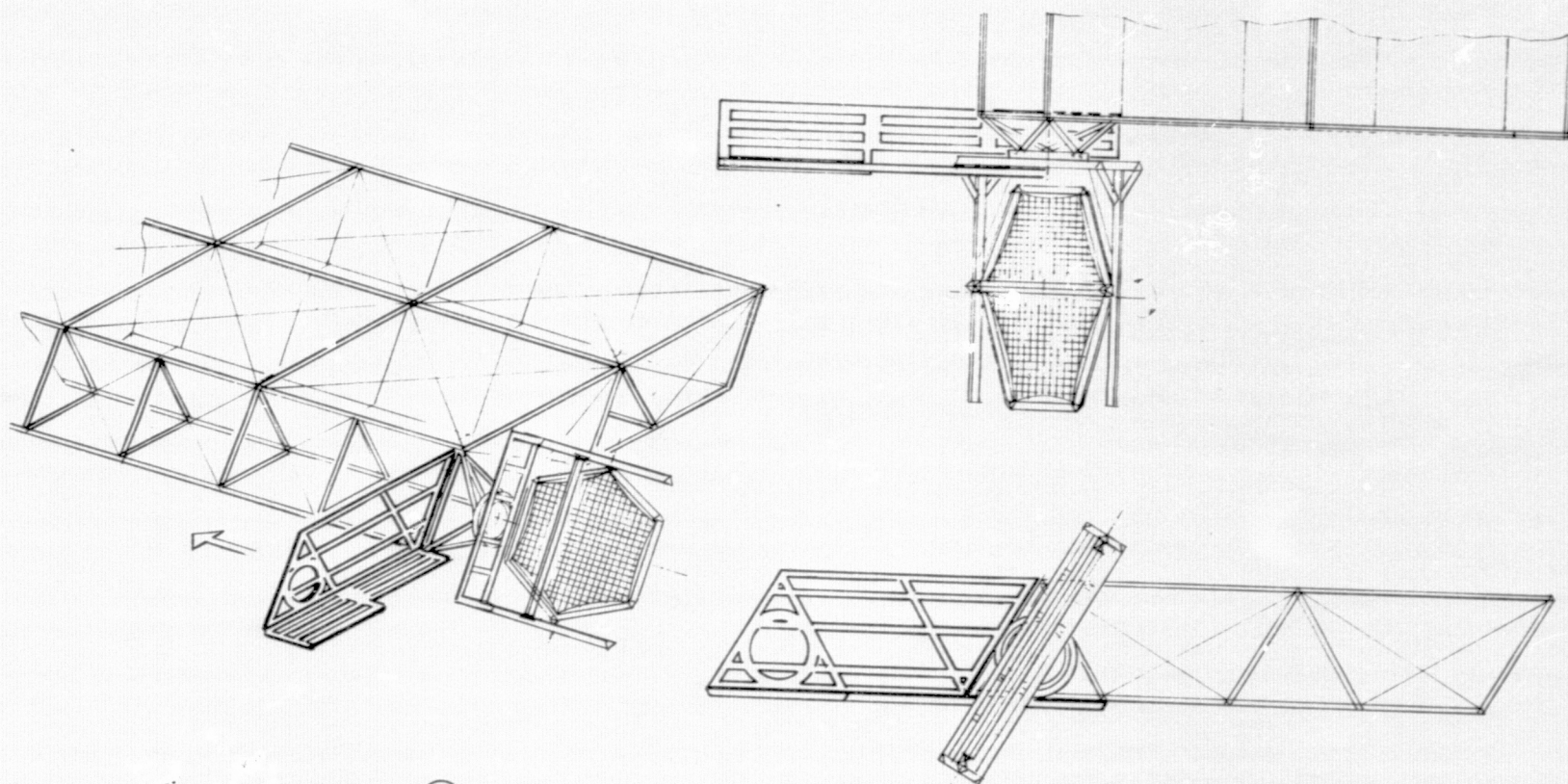
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SCB TRANSLATION FOR TROUGH 3 CONSTRUCTION

The completed antenna installation is shown. At this time the temporary tie bar is removed from the antenna frame and installed in the guideways of the trunnion support arms to become the permanent antenna maintenance platform. Translation of the platform along the guideways, together with the crawler system which transverses its length, provides access to the entire antenna surface. The platform is stored at the base of the yoke during normal satellite operation.

Construction of troughs No. 3 and No. 4 is identical to construction of No's 1 and 2. The completed satellite has been shown previously. After checking out the satellite the SCB is secured and flown away to the site for construction of the next satellite.

SCB TRANSLATION FOR TROUGH 3 CONSTRUCTION



- 9 TEMPORARY TIEBAR TRANSFERRED FROM ANTENNA STRUCTURE
TO TRUNNION SUPPORT BECOMES PERMANENT MAINTENANCE
PLATFORM

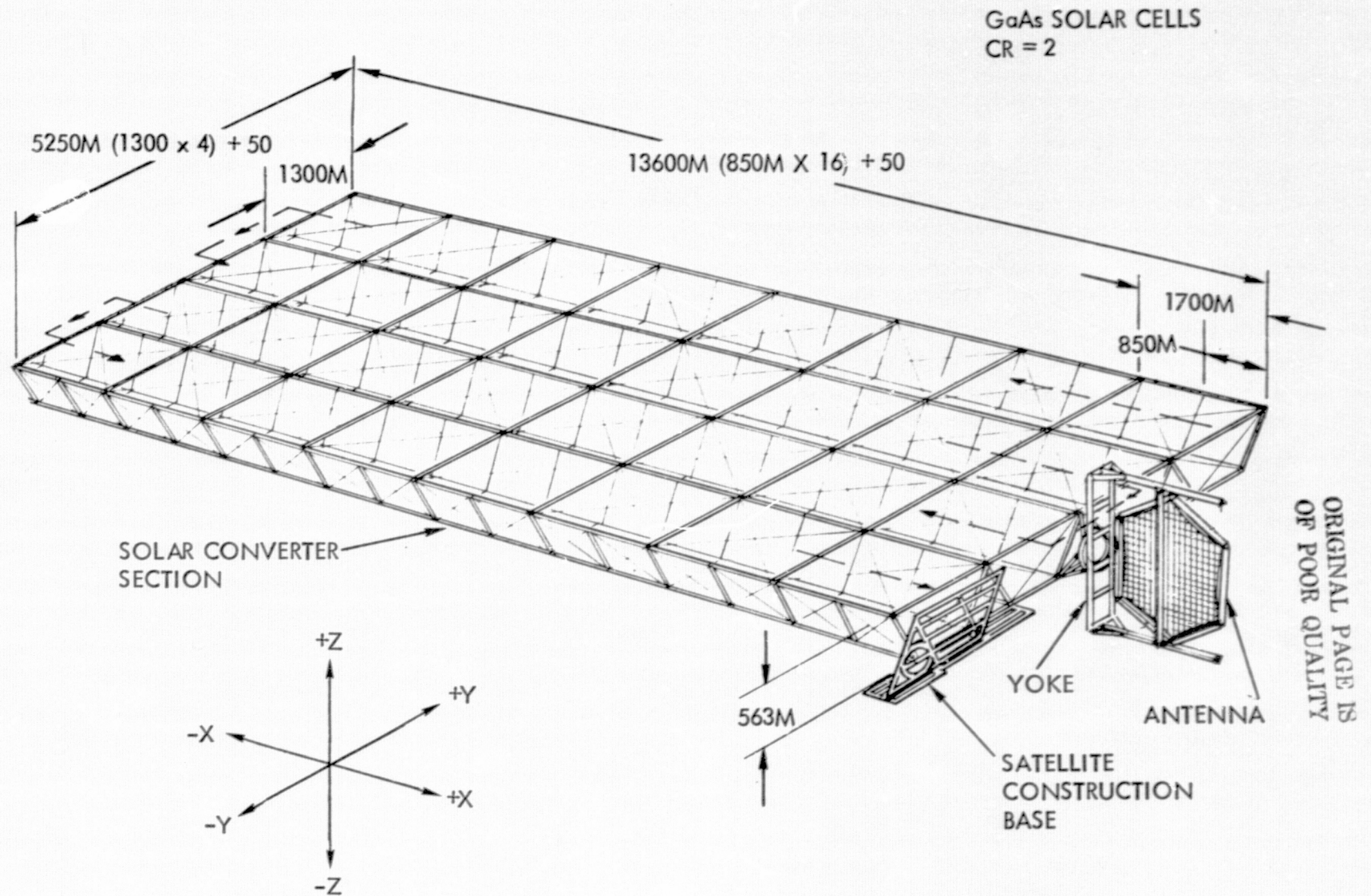
FABRICATION FACILITY TRANSLATES TO NEW POSITION START OF PASS 3.

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COMPLETED SATELLITE

This chart shows the completed satellite. The SCB, which has just finished the fourth pass, will now be detached from the satellite and moved to the next operational satellite location for construction of that satellite.

COMPLETED SATELLITE



CREW SIZE - SERPENTINE CONSTRUCTION

The time-phased crew size for the serpentine construction concept and the 180 day construction schedule previously described is given here. The crew size remains constant during the first 164 days of the construction period at which time the operational maintenance crew (24 members total) arrives on board for their first 90 day tour of duty; during the last 15 days of the construction period the support test and checkout of satellite. Note that the size of the crew is not affected by the number of troughs in the satellite; i.e., a 3-trough satellite would also require a crew size of 264; the required construction time would remain (essentially) the same (180 days) providing the total solar converter area is constant.

CREW SIZE SERPENTINE CONSTRUCTION

OPERATIONS	NO. OF CREW MEMBERS	
	SHIFT SIZE	
	DAY 1-165	DAY 166-180
CONSTRUCTION CREW		
SOLAR CONVERTER CONSTR.	20	20
ANTENNA RF ASSY/INSTLN.	24	24
CONSTRUCTION SUPPORT	6	6
MAINTENANCE	6	6
BASE MGMT	4	4
CREW SUPPORT	6	6
TOTAL SHIFT SIZE	66	66
TOTAL CREW - 4 SHIFTS	264	264
OPERATIONAL MAINTENANCE CREW		24
TOTAL CREW ON-BOARD	264	258

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Rockwell International

Space Division

SCENARIO FOR SINGLE PASS
CONSTRUCTION OF A COPLANAR SATELLITE



Rockwell International
Space Division

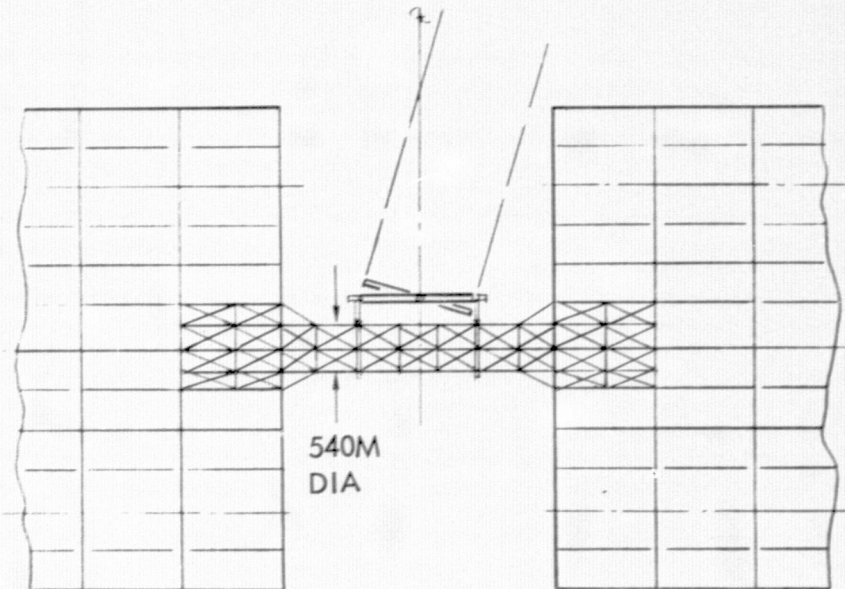
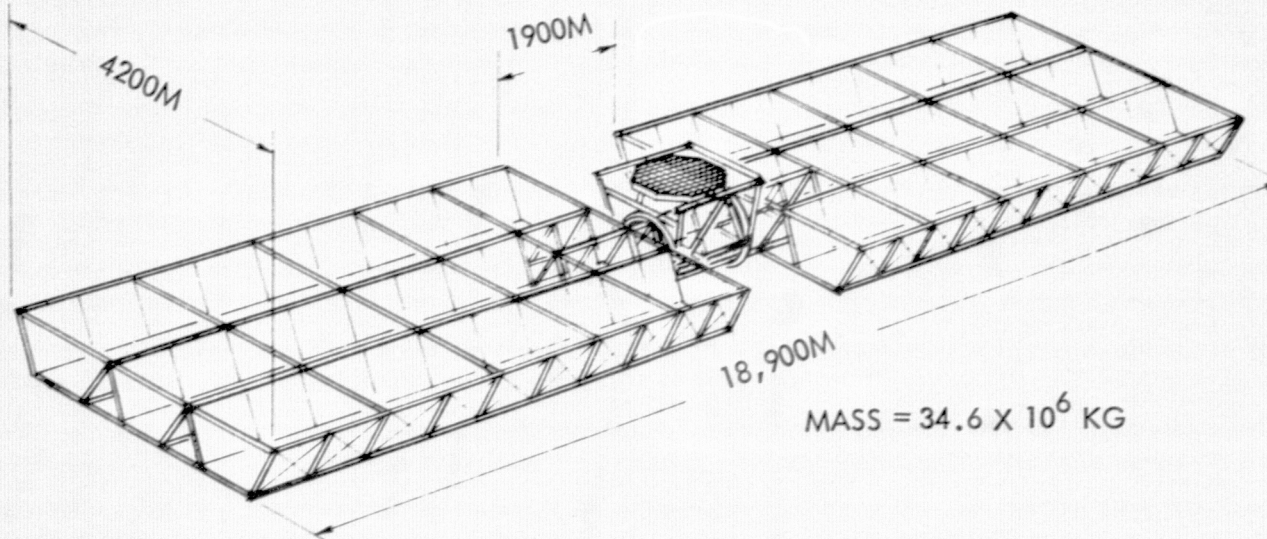
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SATELLITE DESCRIPTION

This 3-trough, center antenna, configuration is the Rockwell recommended configuration for a point-design analysis during the next study period. The overall dimensions, the rotary joint dimensions, and the details of the trough configuration shown on the next chart may be modified somewhat as a result of that analysis but the configuration is sufficiently representative to illustrate the construction scenario presented in the following charts.

SATELLITE DESCRIPTION



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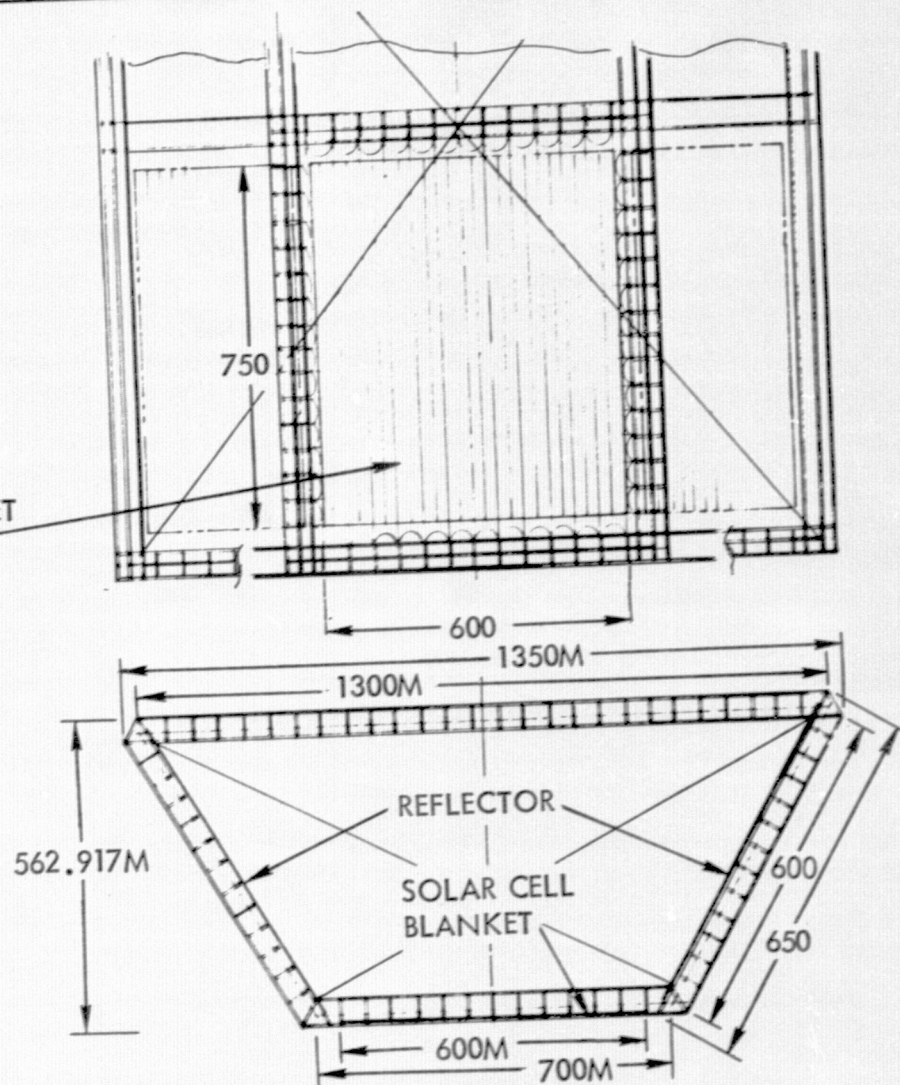
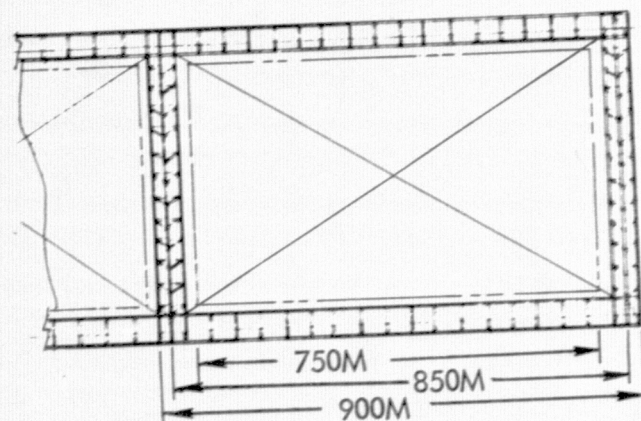
COPLANAR SPS TROUGH CONFIGURATION

The trough configuration for the coplanar satellite with a $CR=2$ consists of a solar blanket area across the bottom of the trough and reflectors installed on the diagonal sides. The dimensions of the trough vary for different satellite configurations, (e.g., three or four troughs) in order to achieve the desired solar blanket area; however, the views shown on the chart are typical.

COPLANER TROUGH CONFIGURATION (REPRESENTATIVE)

GaAs SOLAR CELLS
CR = 2

24 SOLAR BLANKET
STRIPS/BAY



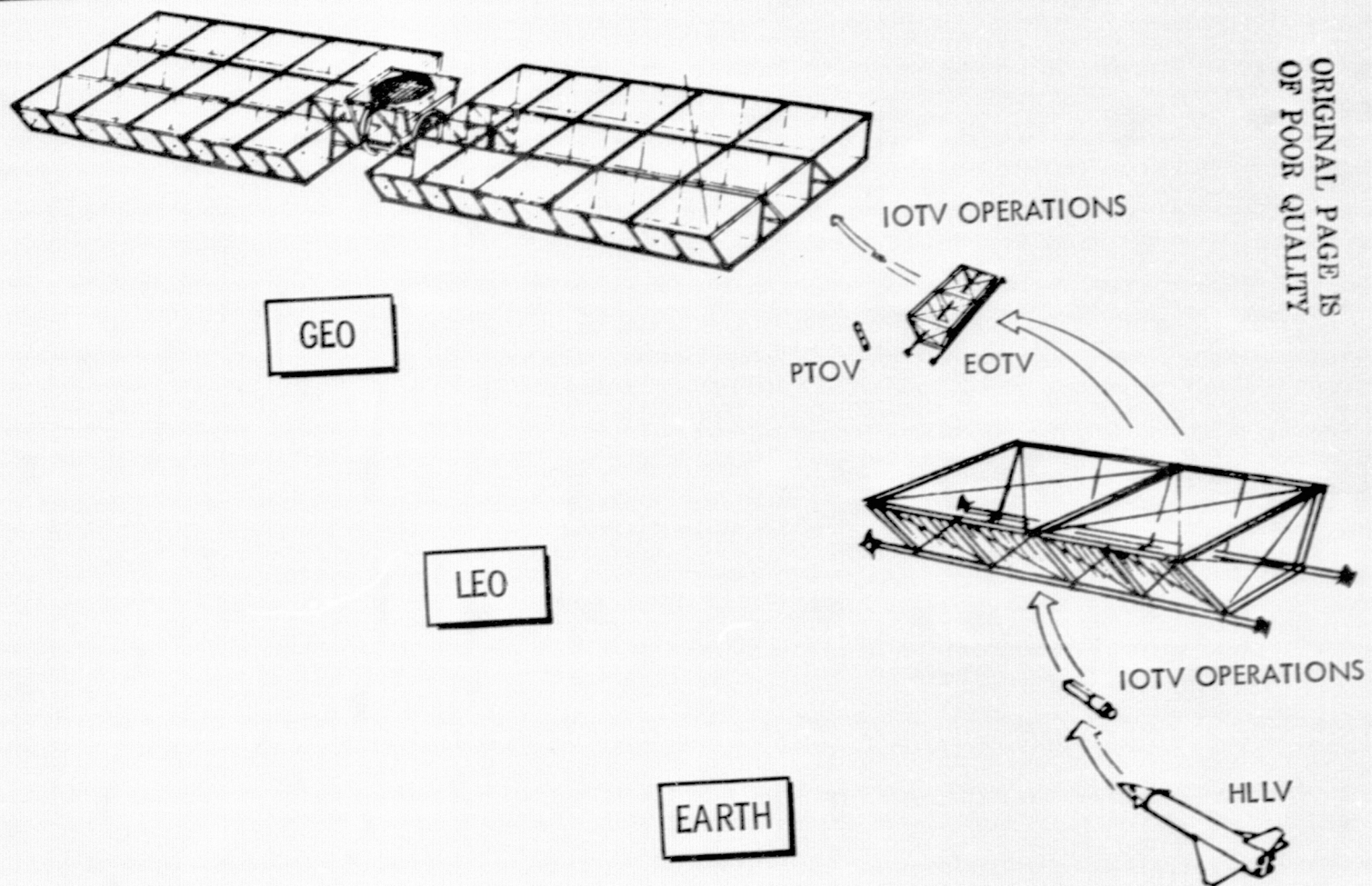
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CONSTRUCTION OVERVIEW

This scenario overview shows the primary operations required to support satellite construction. Initially, construction material is delivered to LEO by HLLV's and transferred directly to EOTV's by the IOTV's. The EOTV's transit to GEO with their cargo, which is similarly transferred to the satellite construction base. The construction crew and their chemically propelled POTV's arrive in LEO via HLLV's, where the stages are mated. The POTV's then proceed to GEO with their crew and are utilized for subsequent crew return to LEO.

"Down" cargo is transferred to LEO by EOTV's. This cargo and POTV's returning from GEO are transported to earth by HLLV's.

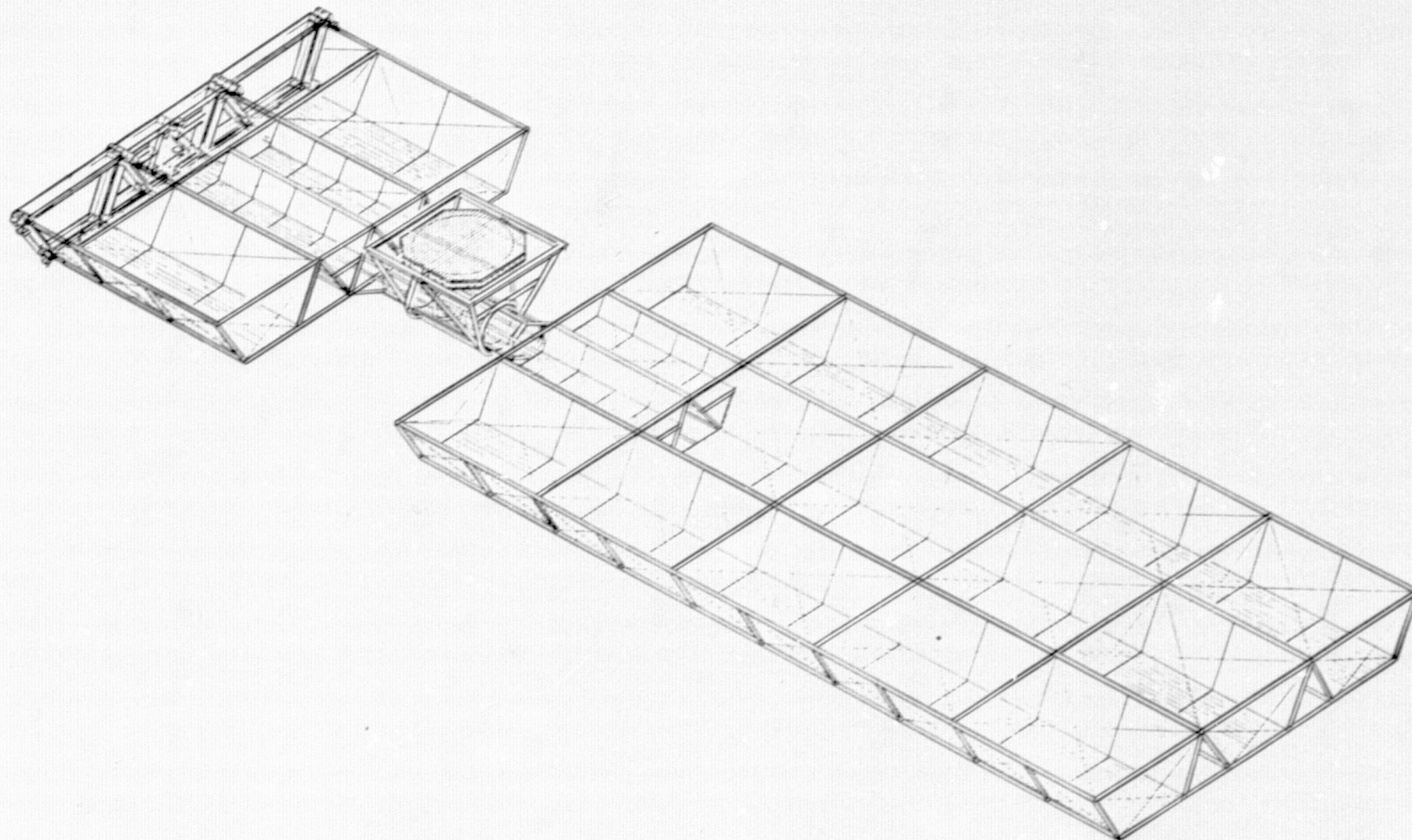
CONSTRUCTION OVERVIEW



SINGLE PASS COPLANAR SATELLITE CONSTRUCTION

This demonstrates the single pass technique. The SCB is on the left end working its way towards completion of wing No. 2. Construction started with the first (right-hand) frame of wing No. 1, constructed the structure and installed the solar converter elements of that wing as it went, and constructed the center section structure including the antenna support structure before moving on to wing No. 2. A separate gantry constructs the antenna structure and installs the RF mechanical modules while wing No. 2 is being constructed. RF subarrays and mechanical modules are assembled in parallel with wing construction operations.

SINGLE PASS COPLANAR SATELLITE CONSTRUCTION



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COPLANAR BASELINE SCB

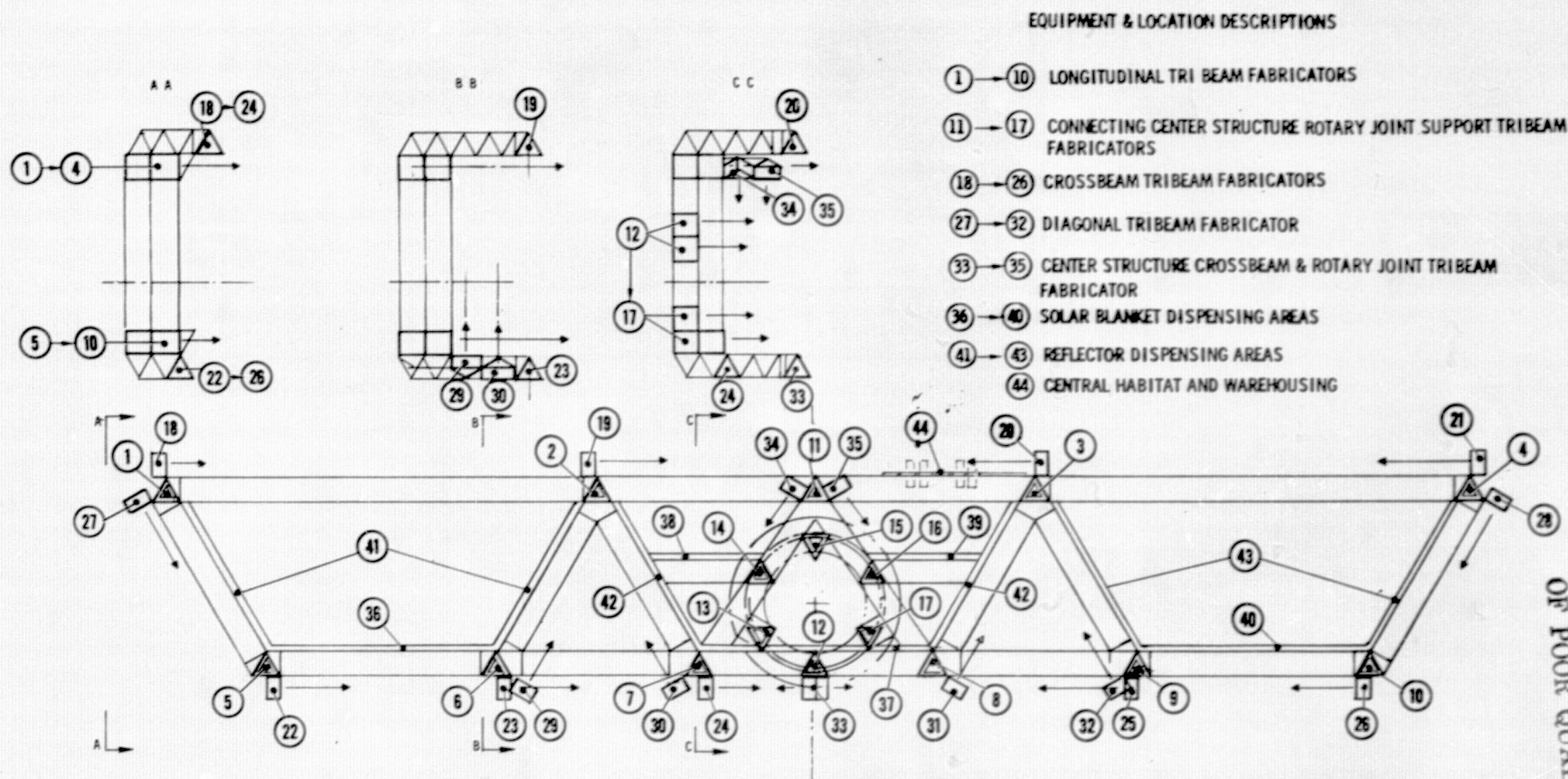
The satellite construction base (SCB) for producing the three trough, center-mounted antenna is shown. The SCB consists basically of three trough fixtures used for simultaneous construction of the three troughs, and a center fixture to fabricate both the rotary joint and the structure which connects the two wings and provides support for the rotary joint.

The solar blankets are installed by means of dispensers which are located along the bottoms of the lower troughs, (36) - (40), and the elevated trough section (38) and (39) (where a short section of elevated solar blanket is required because of interference created by the extension of the center connecting structure into each wing).

The reflector dispensers are located on the diagonals of each trough as designated by (41) - (43).

The crew habitat, power modules, warehousing and docking facilities, (44), are located at the top of the SCB center structure.

COPLANAR BASELINE CONSTRUCTION BASE

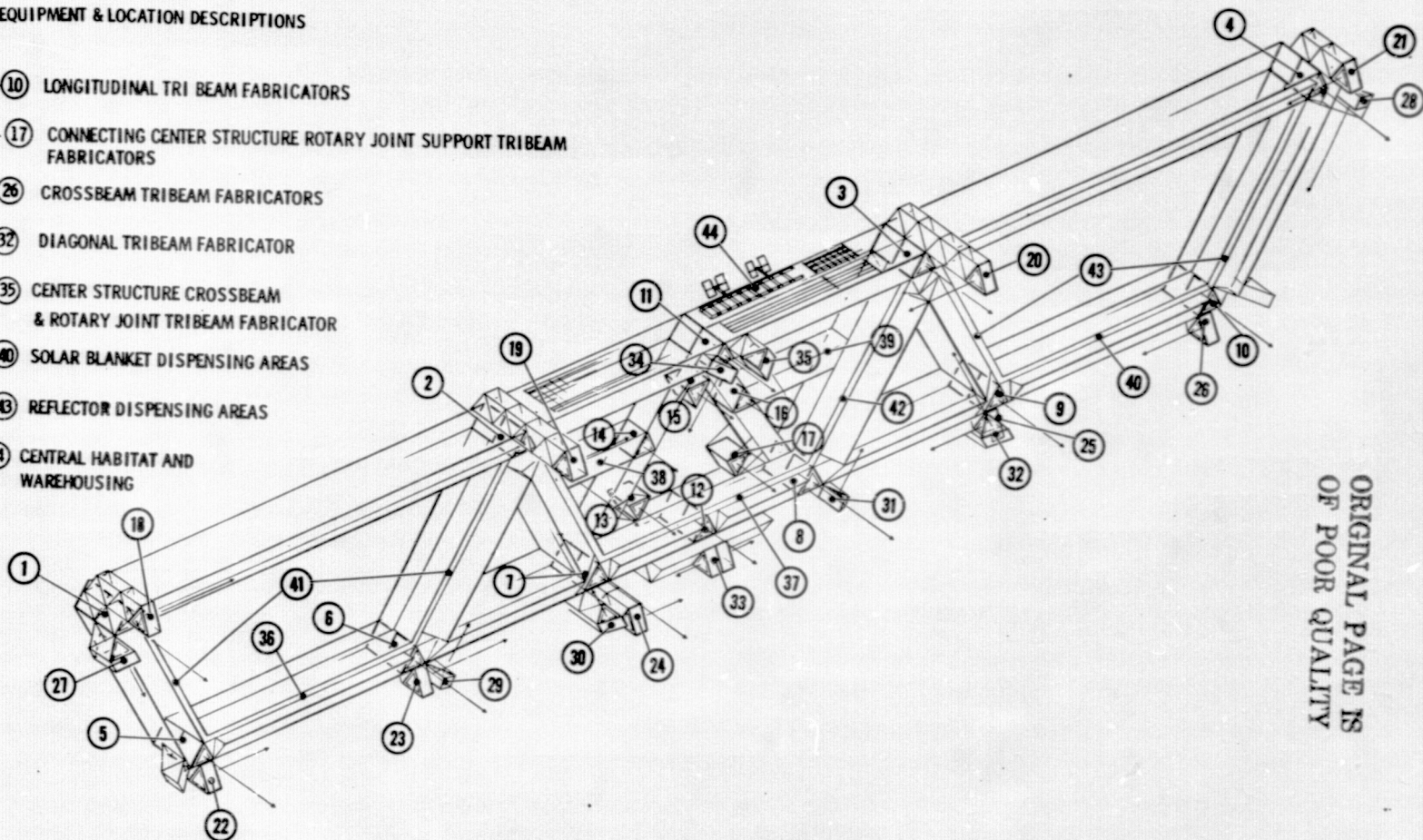


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COPLANAR BASELINE CONSTRUCTION BASE PERSPECTIVE

EQUIPMENT & LOCATION DESCRIPTIONS

- ① → ⑩ LONGITUDINAL TRI BEAM FABRICATORS
- ⑪ → ⑰ CONNECTING CENTER STRUCTURE ROTARY JOINT SUPPORT TRIBEAM FABRICATORS
- ⑱ → ⑳ CROSSBEAM TRIBEAM FABRICATORS
- ㉑ → ㉒ DIAGONAL TRIBEAM FABRICATOR
- ㉓ → ㉔ CENTER STRUCTURE CROSSBEAM & ROTARY JOINT TRIBEAM FABRICATOR
- ㉕ → ㉖ SOLAR BLANKET DISPENSING AREAS
- ㉗ → ㉘ REFLECTOR DISPENSING AREAS
- ㉙ CENTRAL HABITAT AND WAREHOUSING



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SCB MASS SUMMARY
(3 TROUGH - SINGLE PASS)

TRIBEAM FABRICATORS	166000 KG
REFLECTOR INST. EQUIPMENT	120000
SOLAR BLANKET DISPENSORS	187000
(INC. CABLES & CATENARIES)	<u>473000</u>

MICROWAVE ANTENNA	409000 KG
FABRICATION FIXTURE	1499000
LOGISTICS VEHICLES & MANNED MANIPULATORS	<u>252000</u>
	2160000

HABITAT AND POWER SUPPLY	<u>1617000</u>
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TOTAL	4250000 KG
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TOTAL (INC. 25% GROWTH)	5312000 KG
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SATELLITE CONSTRUCTION SCHEDULE (3-TROUGH, SINGLE PASS, SPACE FRAME ANTENNA)

A single integrated construction facility builds the structure, installs the solar blankets, the reflectors, the power distribution system and other subsystem elements located in the wings. Construction starts with one wing tip and progresses toward the center section where the rotating joint for the MW antenna is to be located, and hence continues outboard building wing No. 2 and terminating at the wing tip.

The first eight days are designated for preparation of the construction facility, including distribution and installation into dispensers of material (e.g., structure cassettes, solar blankets, etc.) required to commence construction. During this time satellite materials are arriving from LEO daily with delivery scheduled for completion by the 135th day.

Each satellite wing consists of 10 bays 850-m long. These are constructed at the rate of one every two days using three 8-hour shifts per day. The structure and installation of the power conversion system of wing No. 1 is completed on the 28th day. The center primary structure, rotary joints, antenna support masts, antenna gantry support frame are fabricated during the succeeding 20 days. Utilizing the antenna gantry the antenna primary structure is fabricated in place and its secondary structure installed over the 30-day period from day 46 through day 66. Wing 2 is fabricated over the next 20 day period, completion being on day 86.

Assembly of the RF elements (klystrons and MW radiator panels) into subarrays and mechanical modules for this satellite was initiated on the 145th day of construction of the previous satellite and is continued up through day 86 here.

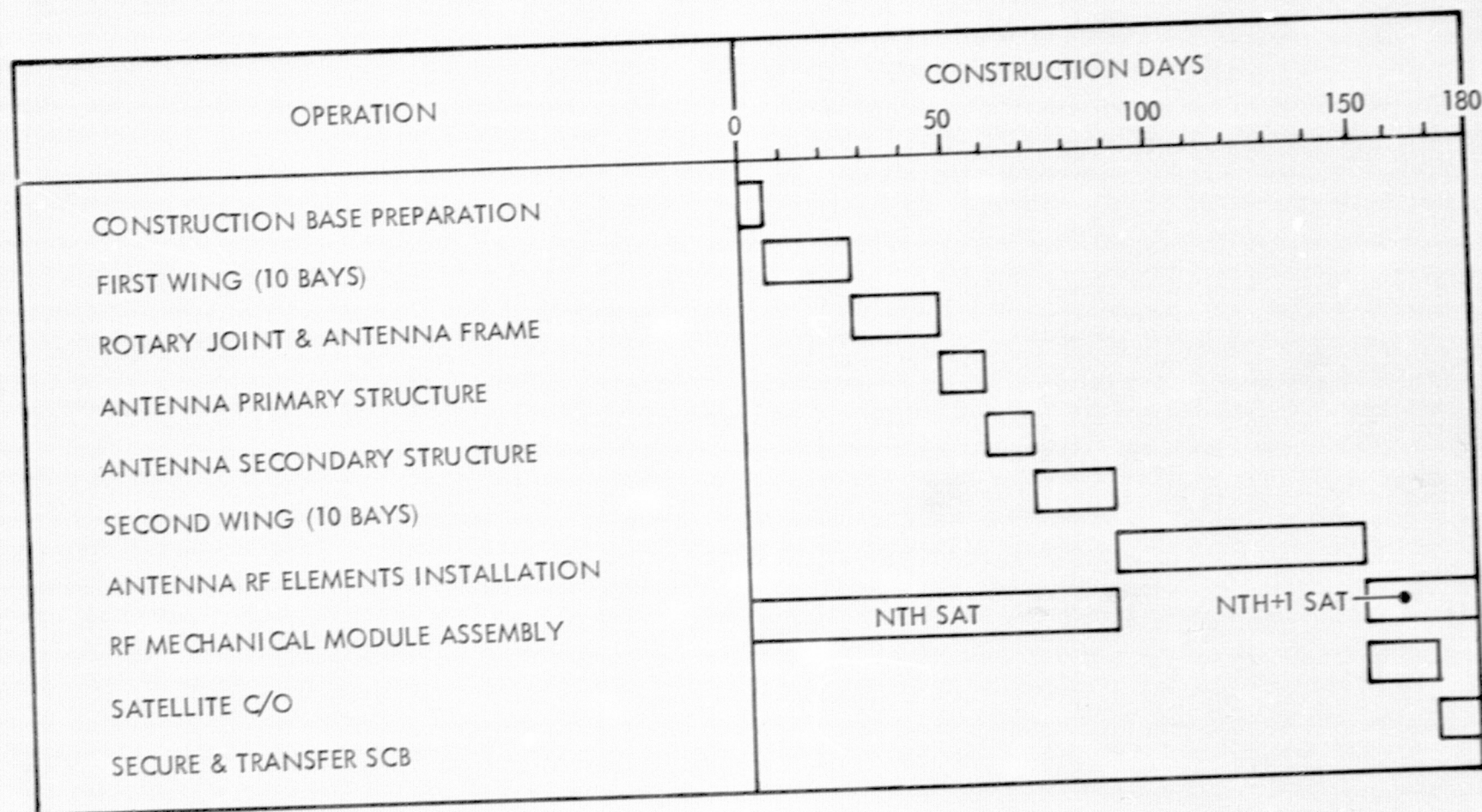
On the 90th day the initial crew's tour is complete. Additionally the structures and solar converter construction has been completed so that only the antenna installation crew is replaced for the basic construction activities during the second 90 day tour.

Installation of the mechanical modules onto the antenna structure commences on the 90th day and continues through the 150th day. At that time the maintenance crew arrives and supports satellite checkout, secure, and SCB transfer operations during the last 30 days of the construction schedule. On the 180th day the construction crew members are returned to earth.

The potential exists for shortening this schedule by increasing the crew size to conduct some of the operations in parallel.

SATELLITE CONSTRUCTION SCHEDULE

(3-TROUGH, CENTER ANTENNA, SINGLE PASS, SPACE FRAME ANTENNA)



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MAJOR SATELLITE CONSTRUCTION ELEMENTS

The major satellite elements and the order in which they will be addressed in this review of the constructability analysis are identified.

MAJOR SATELLITE CONSTRUCTION ELEMENTS

STRUCTURE

{ 2 M BEAMS
50 M TRIBEAMS
INTEGRATED FRAMES & LONGERONS
ACS
CENTER STRUCTURE

SOLAR CONVERTER

{ BLANKETS
REFLECTORS
CONDUCTORS
DATA MGMT & CONTROLS

MW ANTENNA

{ RCR PANELS
KLYSTRONS
MECH MODULES
CONDUCTORS
DATA MGMT & CONTROLS

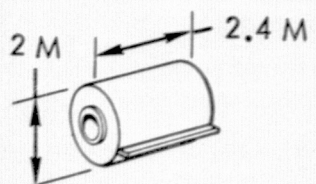
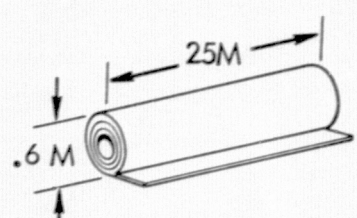
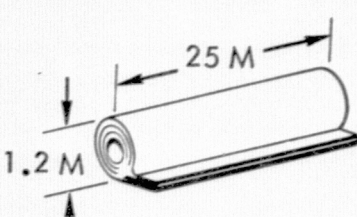
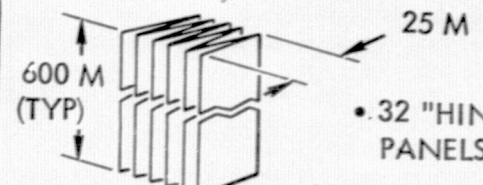
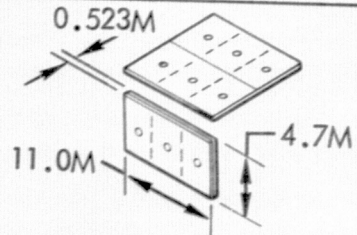


CARGO PACKAGING

These package configurations, sizes and specified quantities required for the construction of each satellite are designed for compatibility with the satellite construction concept and construction equipment. Three primary structure cassettes simultaneously feed each beam machine to produce the basic 2-meter triangular beam elements used in construction of the 50 meter girders. All cassettes contain sufficient length of material to complete one-half of the satellite structure, thus requiring replacement only once during satellite construction.

Each solar blanket roll is 750 m long - the length required for one bay. For a 650 m wide blanket, 26 of these 25 m wide rolls are mounted side by side in the blanket layer and deployed simultaneously.

CARGO PACKAGING

SPS ELEMENT	PACKAGING	PACKAGE DIMENSIONS	NO. REQUIRED	NOTES
STRUCTURES	CASSETTES OF ALUMINUM TAPES		1188	6 DIFFERENT TAPE LENGTHS 2500 KG AVE MASS
SOLAR BLANKETS	ROLLS		1632	750 M LENGTH/ROLL 7136 KG/ROLL
REFLECTORS	ROLLS OF FABRIC-HINGED ALUMINIZED KAPTON SHEET		144	 • 32 "HINGED" PANELS • 12,780 KG/ROLL
MW ANTENNA WAVEGUIDE PANELS	SUB ARRAYS		6993	• ALL SUBARRAYS HAVE SAME OVERALL DIMENSIONS • 10 DIFFERENT POWER MODULE SIZES - QUANTITY VARIES WITH SIZE • SUBARRAY MASS (AVE) = 716 KG

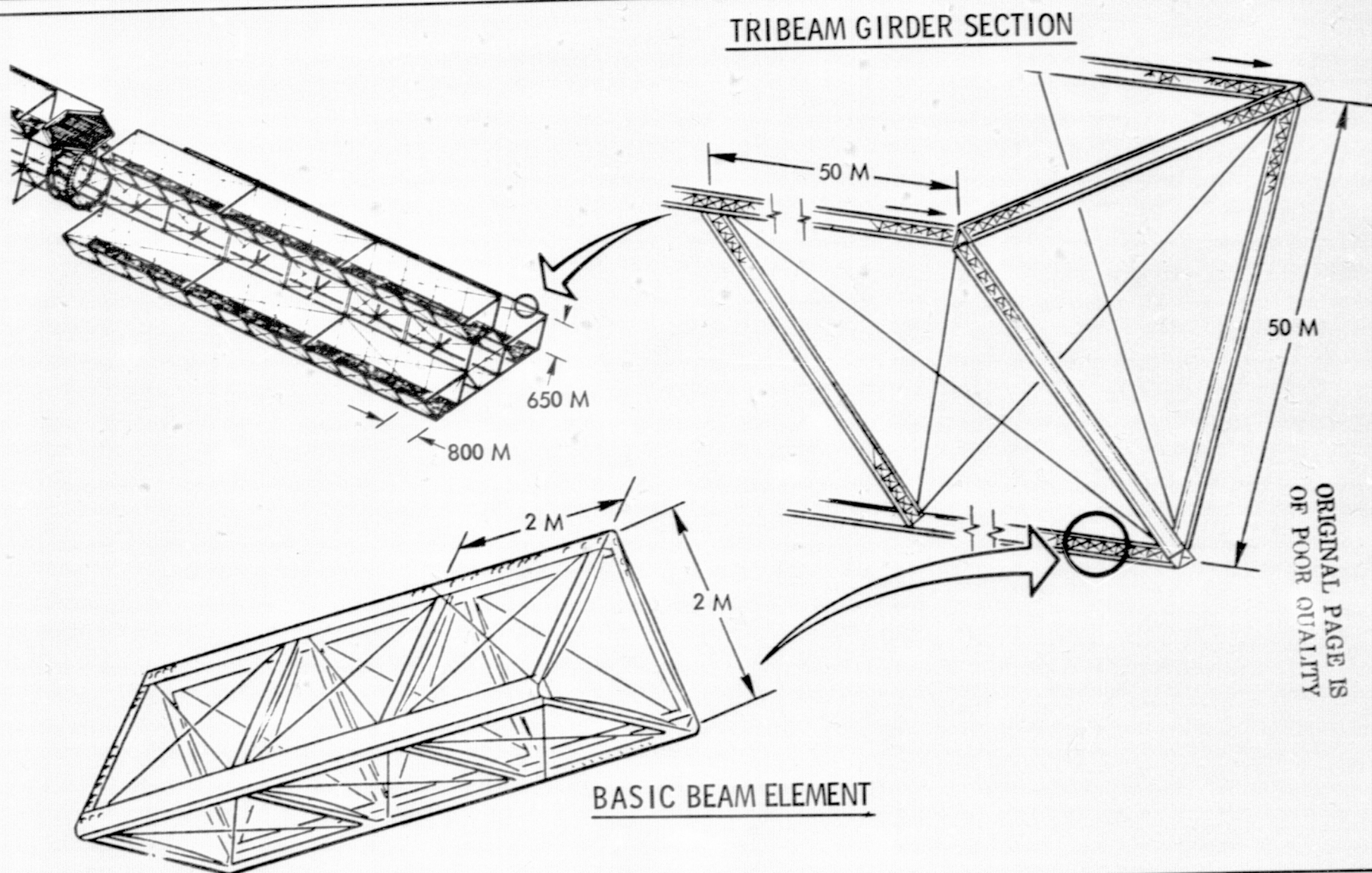
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PRIMARY STRUCTURE EVOLUTION

The two meter tribeam produced by the beam machine is the basic building block for the satellite structure. Its configuration is shown in the chart. The 50-m tribeam is the primary structural element of the satellite and is comprised of two meter tribeams as shown.

PRIMARY STRUCTURE EVOLUTION



BEAM MACHINE

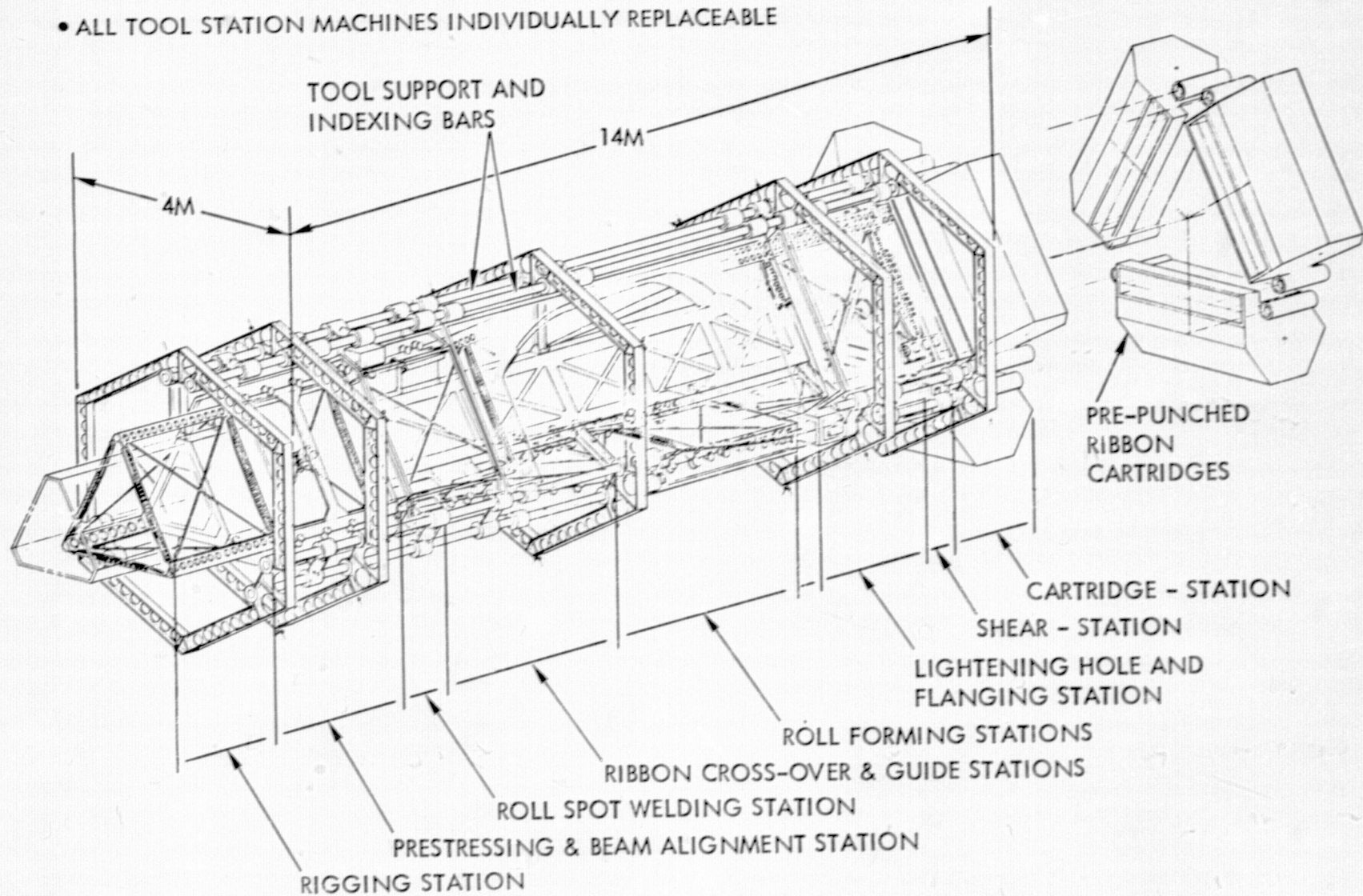
This is a beam machine for making the basic 2-m triangular shaped beams out of aluminum material. The configuration and operation of a machine for making these beams from graphite-fiber composite material will have many similarities to this machine. Each tribeam fabricator requires six beam machines to produce a 50-m tribeam.

Following is a description of the operation of this machine in making aluminum beams:

This machine provides continuous fabrication of the basic 2-m beam elements. Prepunched material transported in cassettes to a structural fabrication facility (to be discussed) and then installed on the supply end of a beam machine. The material in each cassette is automatically threaded into and through the beam machine. Initially, the sheet material enters a shear station where the material rolls are longitudinally indexed. The material next passes into a hole-flanging and struct-forming station. On entry to this station, rolls turn up a small edge which, when passing through the beam fabricator, maintains a cross-load on the sheet when flanged and breake-formed to prevent loss of flat pattern width control. The material then progresses through longitudinal roll forms and is guided through the ribbon cross-over station into a roll seam welder used in a spaced spot-weld mode. The assembled sheet metal element next passes through the prestressing and alignment station where three sheet-metal shrinking-heads shorten and thicken the ribbon cross-braces. Tension-sensing elements control the operation on the basis of preload and alignment requirements fed into the machine from a central computer. The finished triangular shape exits the beam fabricator via a truss-rigging station. Here cable connections, quick-connects, and fittings are installed, when required, that permit the next higher level of assembly by automatic means. Six beam machines are required to produce one 50-meter triangular truss.

BEAM MACHINE

- ALL TOOL STATION MACHINES INDIVIDUALLY REPLACEABLE

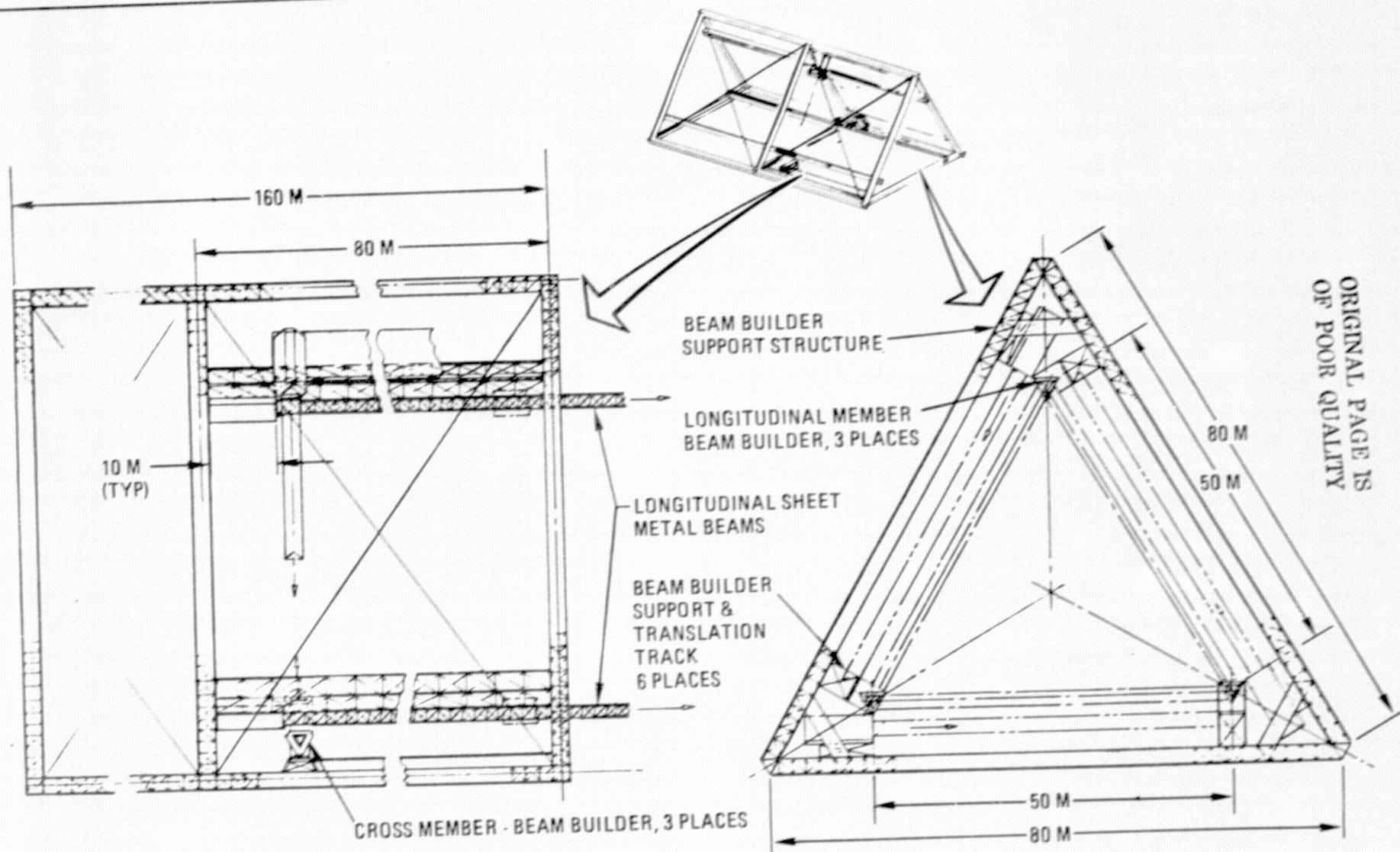


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TRIBEAM FABRICATOR

A housing concept for containment of the six beam machines required to fabricate a 50-meter tribeam is shown. The structure of the tribeam complex would be constructed by beam machines using a more substantial metal thickness, e.g., ~60 mil material. Two tribeam complex lengths are indicated on the left side of the figure. The 160-meter length would be employed for the longitudinal tribeam builders. The added length is used for machine "travel" in the event of malfunction, and an operating rate of 2 meters per minute would allow up to 40-minutes for machine repair before committing to an unscheduled facility shutdown. Since the satellite structural cross-frames are attached to the longitudinal only during scheduled shutdown periods, they need not be designed for "travel" margins.

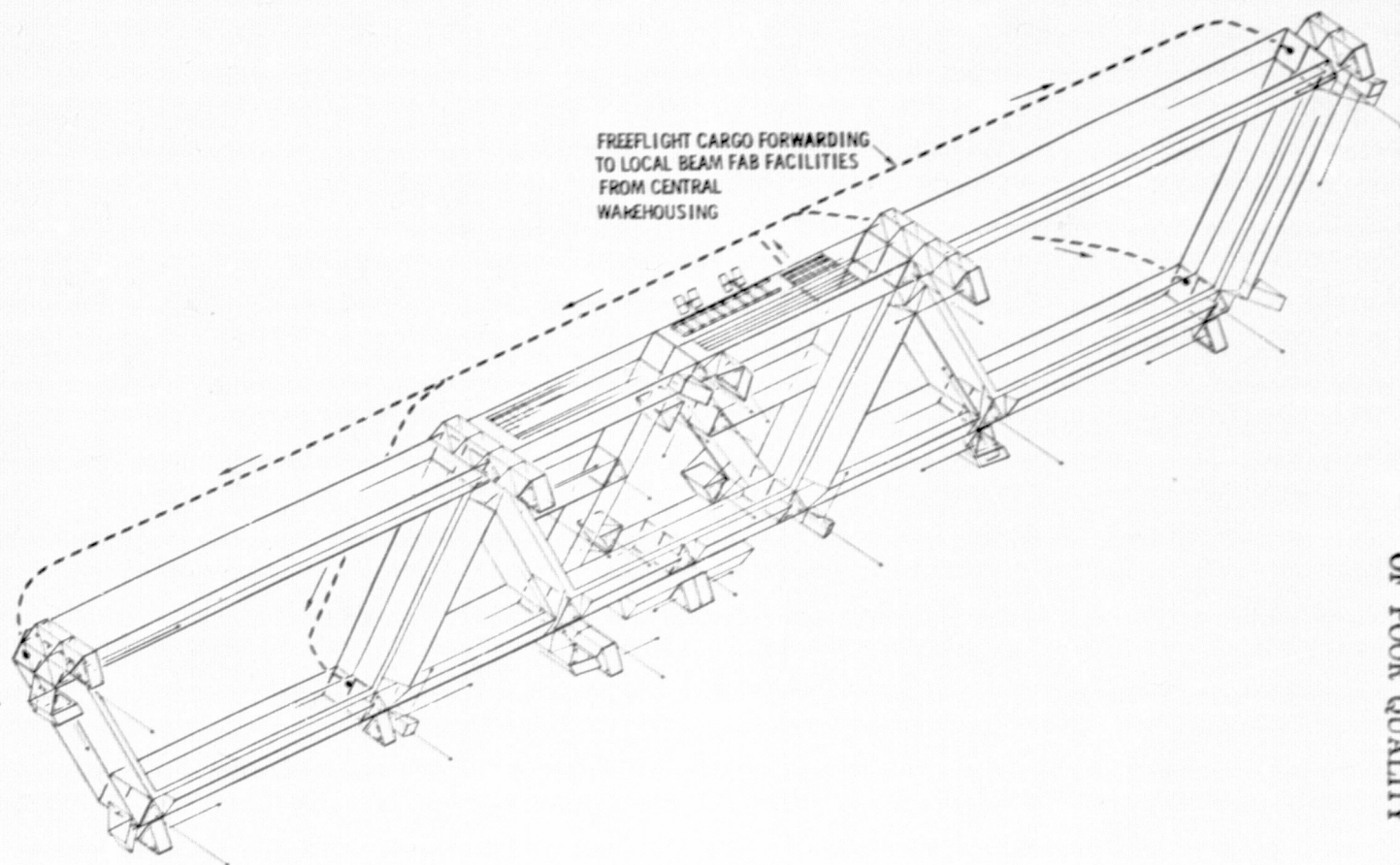
TRIBEAM FABRICATOR



CASSETTE DELIVERY

Aluminum cassettes are delivered by free flying manned logistics vehicles (LV) to the various tribeam fabricating complexes. This chart shows a typical flight path of an LV from the warehouse area to a tri-beam fabricator.

CASSETTE DELIVERY



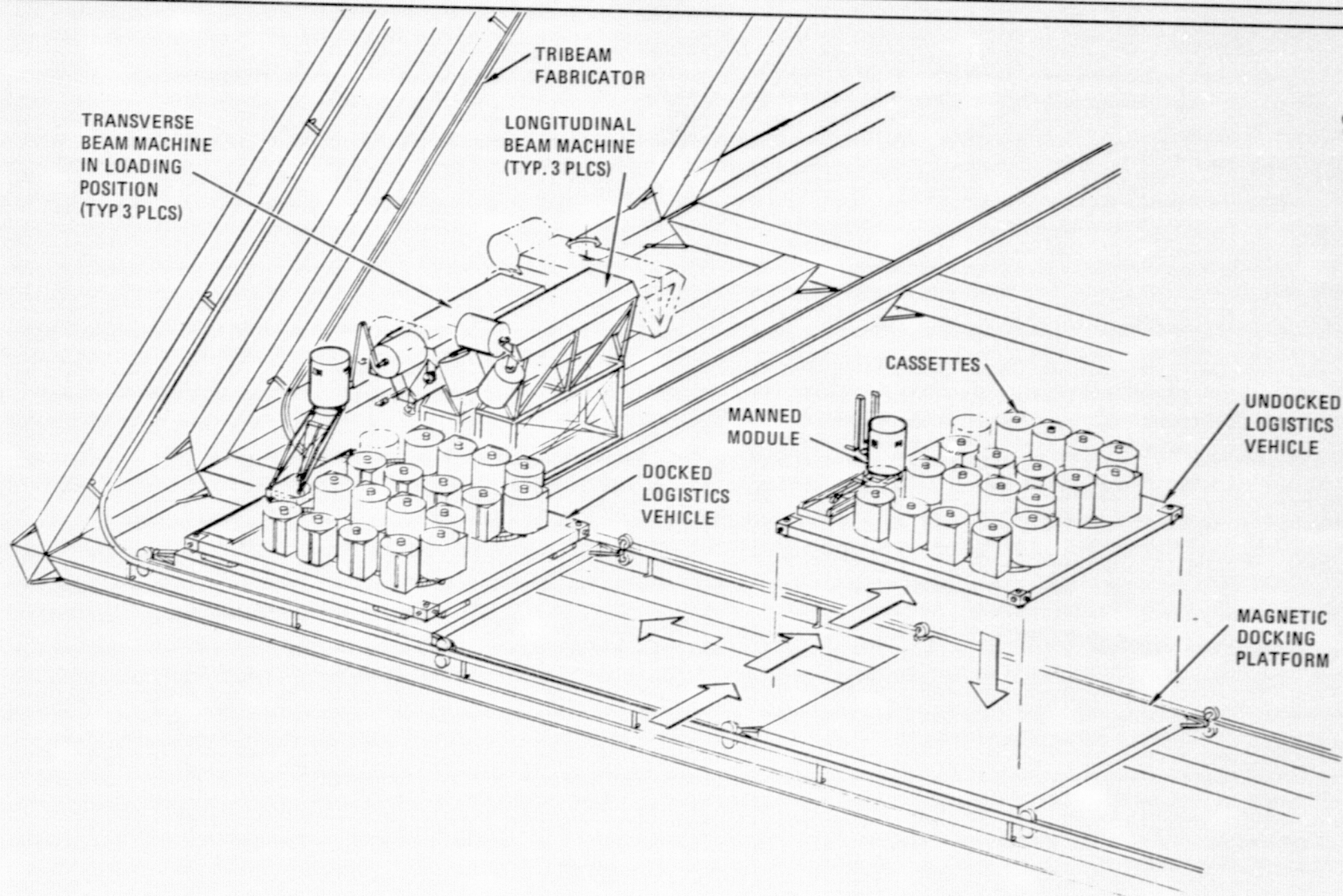
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FREE FLYING LOGISTICS VEHICLE (LV)

This perspective shows the LV in both the undocked and docked position. In the docked view, the manned module is in the process of removing an empty cassette for stowage on the platform prior to replacing it with a full cassette. The cassette is being removed from a transverse beam machine which has been rotated and translated into servicing position. The tracks to which the docking platform is attached are designed to permit travel to all three corners of the tribeam fabricator.

FREE FLYING LOGISTICS VEHICLE (LV)

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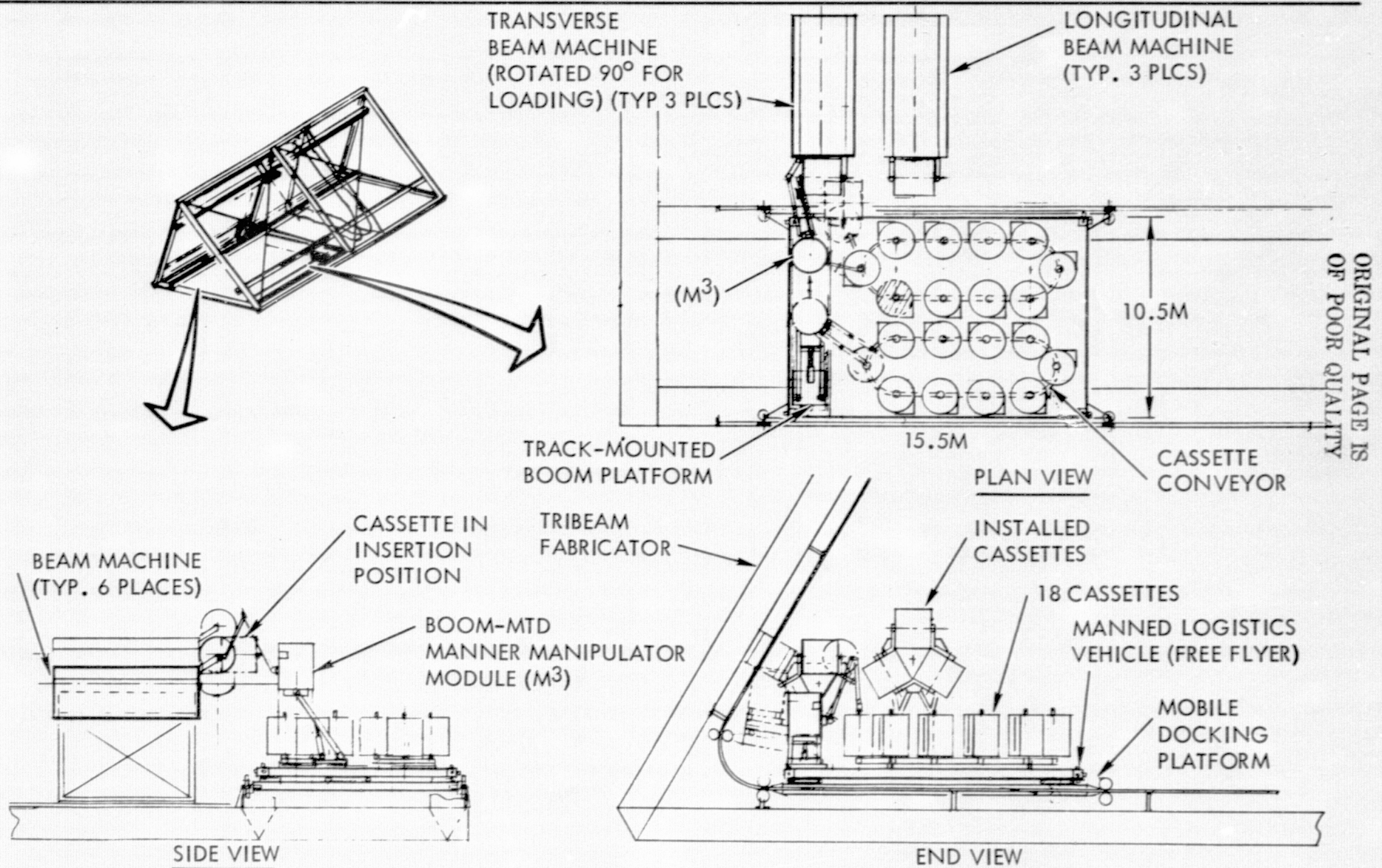


TRIBEAM FABRICATOR SERVICING CONCEPT

There are 33 tribeam fabricators utilized for satellite construction. These fabricators are installed in 14 general locations. Each fabricator must be loaded with 18 aluminum cassettes (3 per each of 6 beam machines) twice to support construction of one satellite, since the cassettes are sized to provide material for one wing only. The cassettes are transported from the warehouse area of the SCB to the various locations. This can be done either by a vehicle traveling on tracks or cables, or by a free flier. The network of tracks (or cables) which would be required for servicing the various locations from the warehouse area is complex and involves numerous changes of direction, some of which would be difficult to traverse. For these reasons, the free flying mode was selected. The concept consists of a chemically propelled and stabilized manned logistics vehicle (LV) capable of transporting 18 cassettes. The cassettes are attached to a conveyor on a detachable flatbed, which permits preparation of one load while the other is being delivered. The loaded LV docks on a track-mounted magnetic docking pad located at the rear of the fabricator. The tracks traverse the three sides of the fabricator, thus providing access by the LV to the six beam machines. Each empty cassette, mounted in a swivel hub and secured at the other end by a yoke arrangement, is removed and replaced by a full cassette. It is noted that the beam machines which construct the cross beams rotate and translate into position parallel to the longitudinal machines to facilitate unloading and loading.

The initial loading operation is conducted during preparation of the SCB for the next satellite construction. The second operation is conducted following completion of the first wing. Sufficient LV's are available to permit accomplishment of the 8 hour operation at the various stations within the overall construction time line. LV propellant requirements are minimal, amounting to approximately 850 kg per sortie.

TRIBEAM FABRICATOR SERVICING CONCEPT



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TRIBEAM FABRICATOR (TBF) SERVICING TIMELINE

The time required for major activities associated with servicing a tribeam fabricator is indicated. A total time of eight hours is required for servicing each TBF; however, servicing is required only twice for construction of a satellite with aluminum primary structure.

TRIBEAM FABRICATOR (TBF) SERVICING TIMELINE

● LOAD LOGISTICS VEHICLE (LV) AT WAREHOUSE	
● DEPART WAREHOUSE FOR FABRICATOR, DOCK.	0.5
● CONNECT LV TO TBF CONTROL BOX	0.1
● POSITION BEAM MACHINES	0.1
● POSITION LV (3 TIMES @ 0.1 EACH)	0.3
● UNLOAD/LOAD CASSETTES (6 LOCATIONS)	6.0
● UNLOAD/LOAD FITTING MAGAZINE	0.4
● POSITION LV FOR DEPARTURE	0.1
● RETURN TO WAREHOUSE	0.5
	8.0 HRS

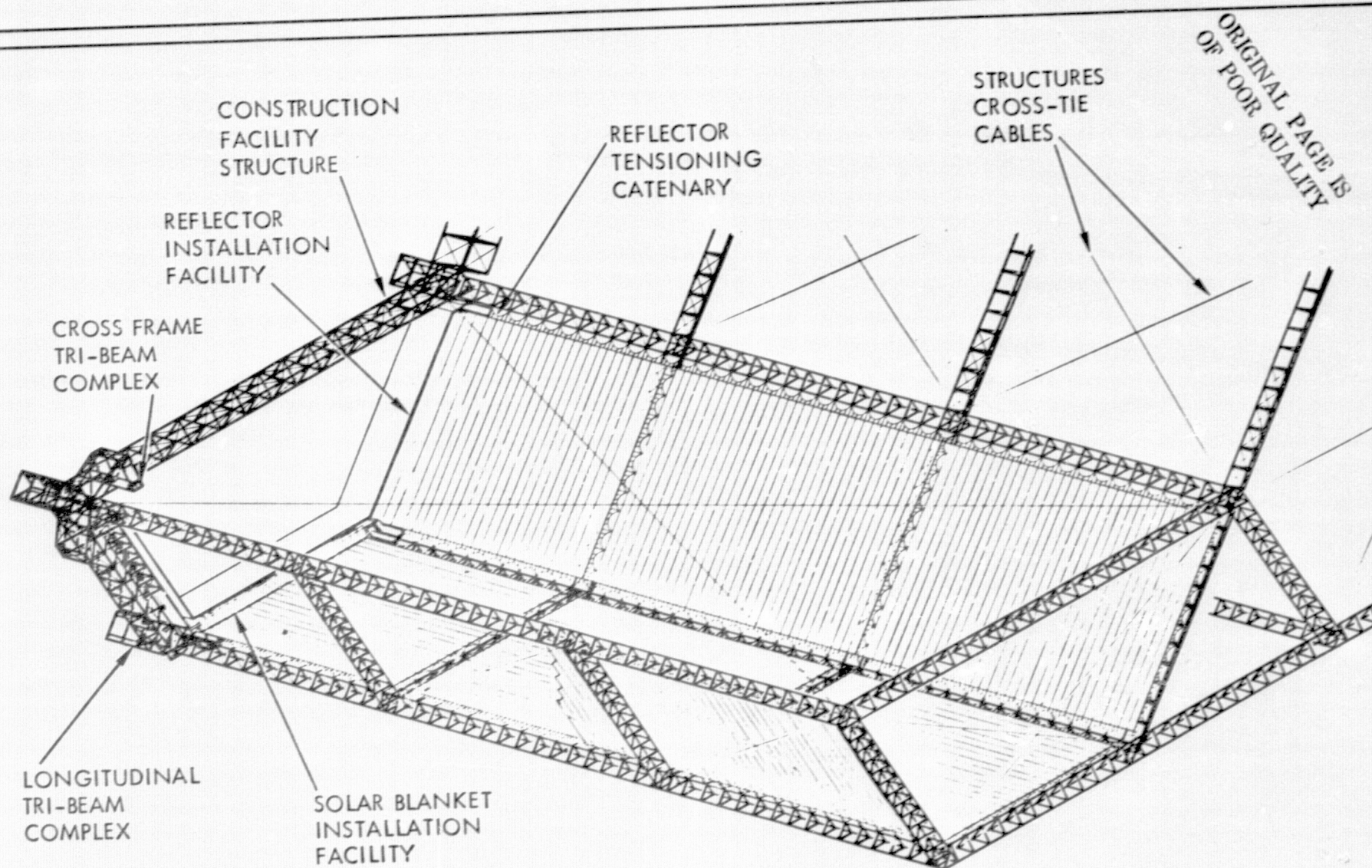


SOLAR CONVERTER CONSTRUCTION

(Structures, Solar Blankets and Reflectors)

The perspective drawing illustrates the near-completion of the first three 800-meter "bays" in the lower corner of the satellite with a section of the outside reflector panels cutaway. It can be seen that the solar blankets are laid out in horizontal strips but that the reflector panels are vertically oriented. The structure of an 800-meter bay is estimated to take one 8-hour shift to fabricate. During this time, the solar blankets are "played out" - from 25-meter rolls - and edge-attached to longitudinal lines of composite materials; the reflectors are refurled (to be shown later) and also loosely constrained by vertical lines. Upon reaching the end of a bay, the construction facility is stopped and, during the next five 8-hour shifts, the cross frame members are attached, the solar blankets are secured and the reflector panels are tensioned.

SOLAR CONVERTER CONSTRUCTION (STRUCTURES, SOLAR BLANKET & REFLECTORS)

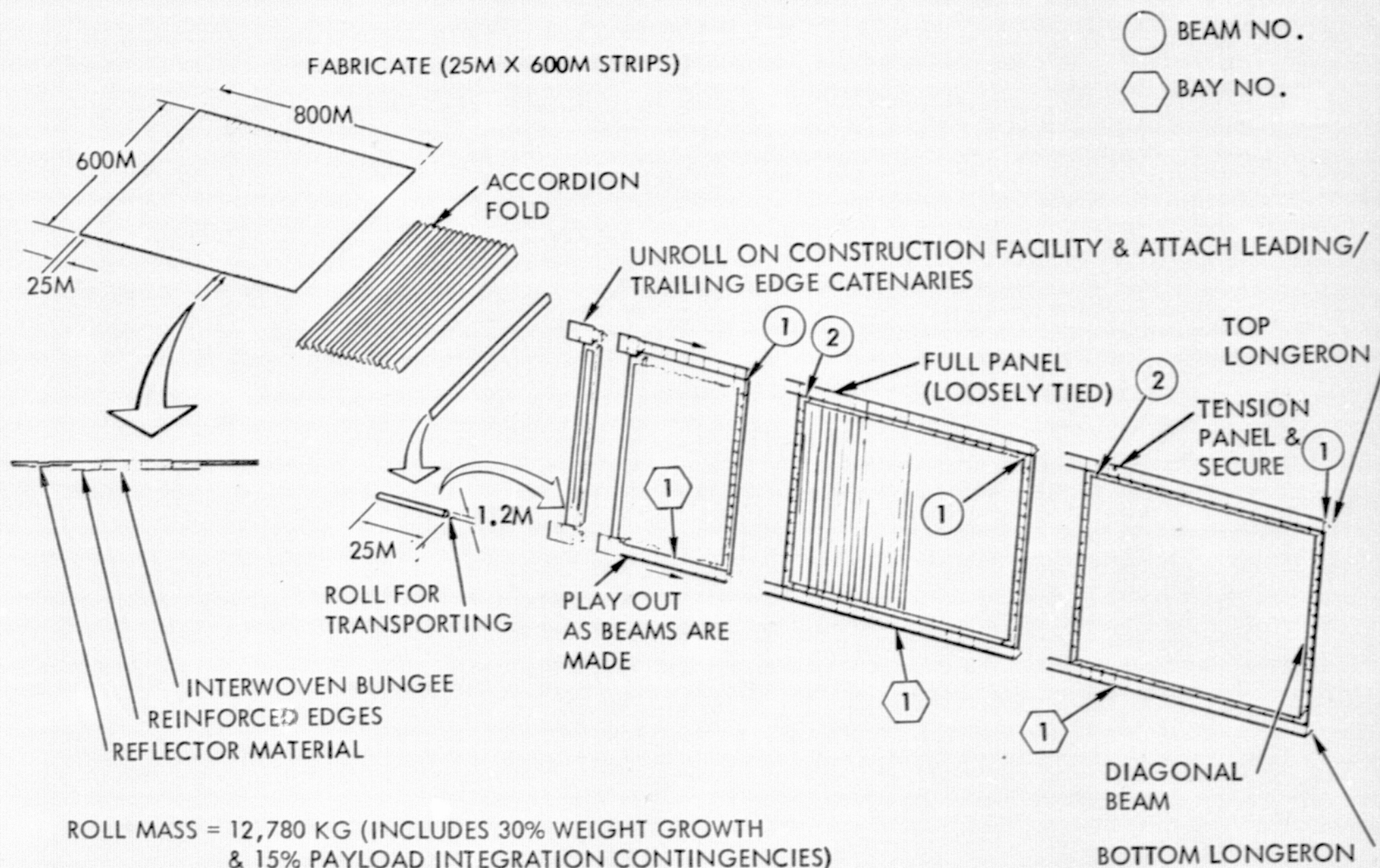


REFLECTOR PACKAGING AND INSTALLATION

The reflector panels, measuring 600-m $\times$ 800-m, are pleated at 25-m intervals to produce an accordian type fold as shown. They are then rolled along the plane of the end pleat into a roll 25-m long and 1.2-m diameter which is the configuration for transporting into orbit.

When installed, each reflector panel is suspended within the 800-m bay by longitudinal catenaries attached to the upper and lower longerons and by leading and trailing edge catenaries attached to the forward and aft diagonal members of the transverse frames. The catenaries are attached to the trailing and leading diagonal transverse beams and to the longerons. Two panels are required for each 800-m bay of each trough or a total of 144 panels for the entire satellite.

REFLECTOR PACKAGING & INSTALLATION



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ROLL MASS = 12,780 KG (INCLUDES 30% WEIGHT GROWTH
& 15% PAYLOAD INTEGRATION CONTINGENCIES)

NO. ROLLS = 144

SOLAR BLANKET INSTALLATION CONCEPT

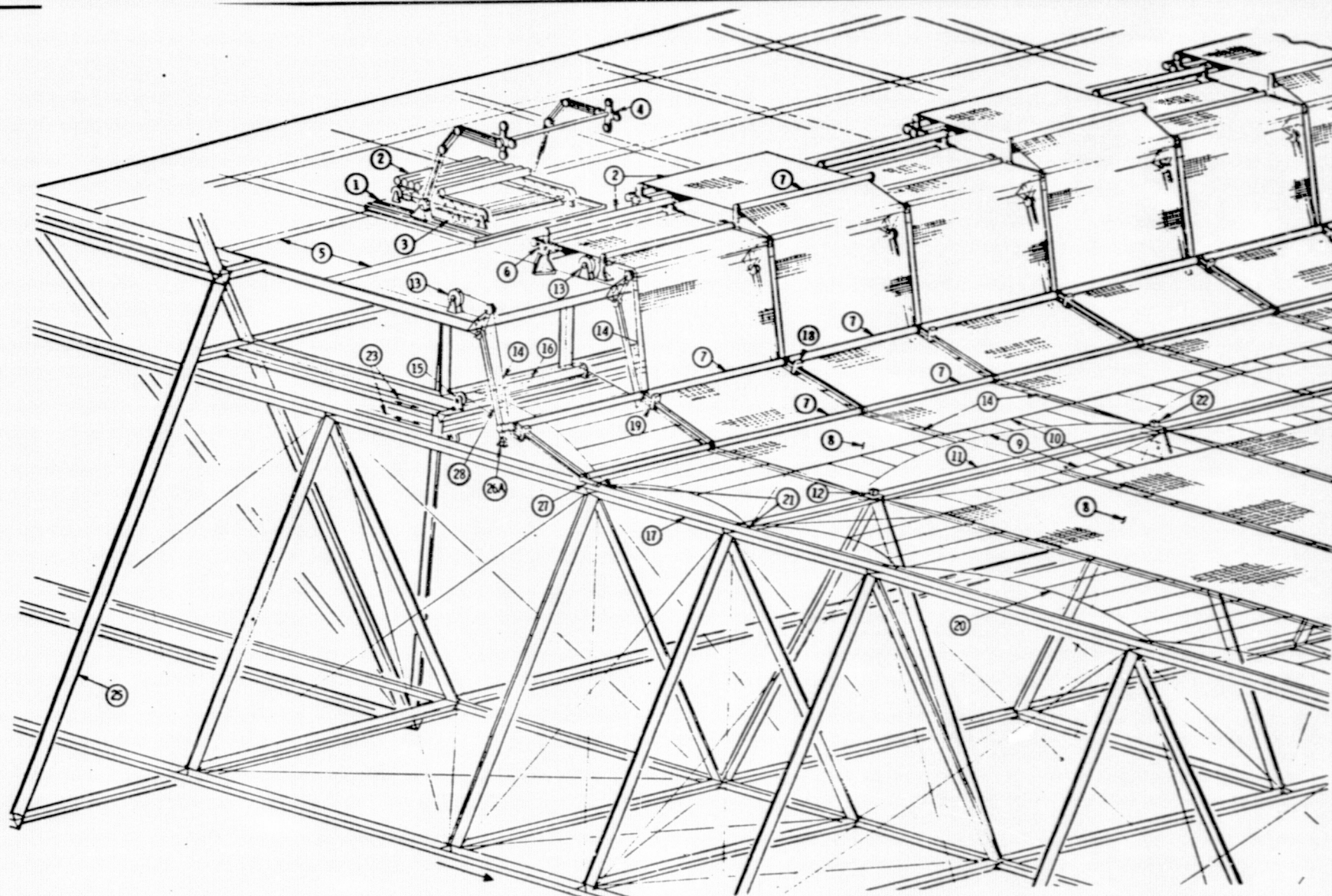
(The callouts indicated by the circled numbers are given on the two pages following the figure.)

The solar blanket in each 800 m long bay is a structurally independent installation suspended by side and end catenaries attached to the longerons and cross beams respectively, and by longitudinal cables stretched between the blanket strips. Each blanket strip is approximately 25 m wide and 750 m long, and is packaged in a 25 m wide roll by 0.6 m in diameter. Each two bays of solar blankets are electrically connected in series, constituting a functional module which produces the required voltage.

Initially the blanket rolls are transported from the SCB warehouse area by a transporter/loader (1) which inserts the rolls into the dispensers (6). The leading edge of the blanket strips with end catenaries attached, are then threaded through the roller arrangement and attached to the trailing edge of the cross beam just completed. The longitudinal cables to which the side edges of the blanket will be fastened are threaded from the cable dispenser (13) and attached in a similar manner. The longitudinal catenaries are fabricated on the middle deck, fed into the dispensing spindle (15) and then attached to the cross beam trailing edge.

Solar blankets and catenaries are attached to the longitudinal cables by foldover tabs which are applied by automatic fastening equipment. As the cross beam advances the blanket strips, longitudinal catenaries and cables are payed out. The two outside cables are attached to the longitudinal catenaries, the two longitudinal catenaries to their respective longerons, and the inside edges of adjacent blanket strips to their stabilizing cables. Upon completion of the bay and the next following cross beam, the trailing edges of the blankets (i.e., the trailing transverse catenaries) and the trailing end of the longitudinal catenaries are attached to the leading edge of that cross beam. The installation is then tensioned and electrical connections completed.

SOLAR CELL BLANKET INSTALLATION CONCEPT



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EQUIPMENT AND LOCATION DESCRIPTIONS

- ① S/A BLANKET ROLL TRANSPORTER - LOADER
- ② S/A BLANKET ROLLS
- ③ USED ROLL CORES
- ④ BLANKET ROLL INSTALLER - REMOVER
- ⑤ TRANSPORTER TRACKS
- ⑥ S/A BLANKET DISPENSING SPINDEL QUAD
- ⑦ BLANKET STRIP GUIDE ROLLERS
- ⑧ DEPLOYED SOLAR BLANKET STRIP
- ⑨ LEADING TRANSVERSE CATENARY
- ⑩ BLANKET STRIP - TRANSVERSE CATENARY JOINT LINE
- ⑪ UPPER VERTEX OF SATELLITE (50 M TRIBEAM GIRDER) CROSS BEAM
- ⑫ TRANSVERSE-CATENARY-TO-CROSS BEAM ATTACH POINT
- ⑬ LONGITUDINAL CABLE DISPENSER
- ⑭ LONGITUDINAL CABLE
- ⑮ LONGITUDINAL CATENARY DISPENSING SPINDEL
- ⑯ LONGITUDINAL CATENARY ROLL
- ⑰ UPPER VERTEX OF SATELLITE (50 M TRIBEAM GIRDER) LONGERON IN
BOTTOM CORNER OF TROUGH
- ⑱ BLANKET-EDGE-TO-CABLE ATTACH MACHINE
- ⑲ BLANKET-EDGE-TO-LONGITUDINAL CATENARY ATTACH MACHINE
- ⑳ DEPLOYED LONGITUDINAL CATENARY
- ㉑ CATENARY-TO-LONGERON ATTACH POINT
- ㉒ SWITCH GEAR MOUNTED ON CROSS BEAM

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EQUIPMENT AND LOCATION DESCRIPTIONS (CONT.)

- ②③ RETRACTING PLATFORM FOR SWITCH GEAR AND SECONDARY FEEDER INSTALLATION
- ②④ MAIN FEEDER DISPENSER
- ②⑤ CONSTRUCTION FIXTURE
- ②⑥ TRANSVERSE CATENARY-TO-CROSS BEAM ATTACH MACHINE IN ATTACH POSITION; 26A NON-ATTACH POSITION
- ②⑦ LONGITUDINAL CATENARY-TO-LONGERON ATTACH MACHINE IN ATTACH POSITION; 27A NON-ATTACH POSITION
- ②⑧ ATTACH EQUIPMENT TRANSLATING SUPPORT ARM
- ②⑧A TRANSLATING ARM IN CATENARY-TO-CROSS BEAM ATTACH POSITION
- ②⑨ CROSS BEAM (50 M TRIBEAM GIRDER) FABRICATION FACILITY IN BEAM FABRICATION POSITION
- ②⑨A 50 M CROSS BEAM IN FABRICATION POSITION



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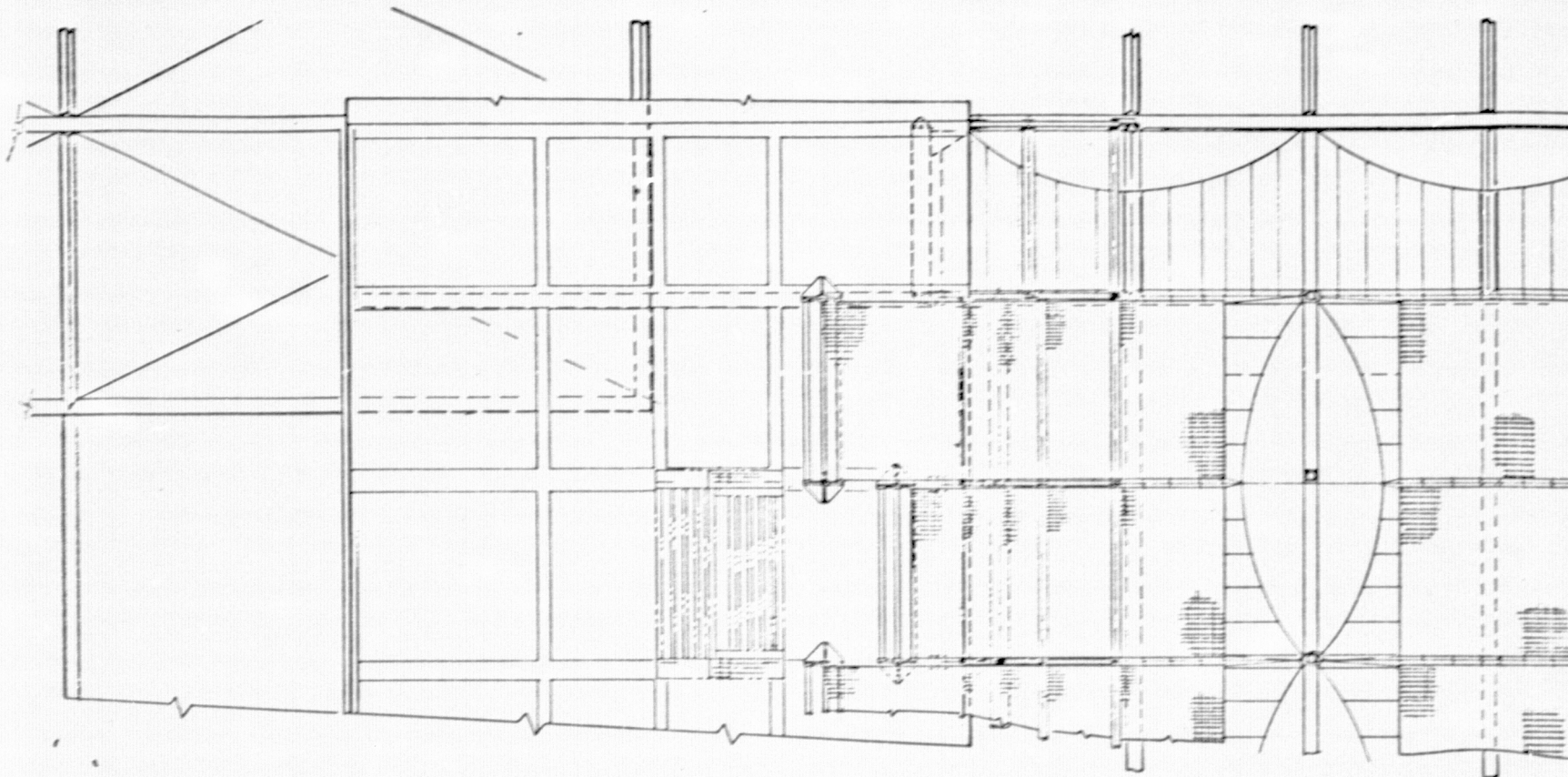
Space Division

SOLAR BLANKET STATION PLAN

This is a (partial) plan view of the upper deck solar blanket installation station. It shows the transporter/loader in position to service a dispenser. The dispensers are staggered to prevent interference between adjacent stations.

A leading edge and trailing edge catenary is shown in the right portion of the chart. A single 50 m wide catenary connects each two blanket strips to the cross beam. The longitudinal catenary which provides attachment and tensioning to the longerons also is shown.

BLANKET INSTALLATION STATION PLAN



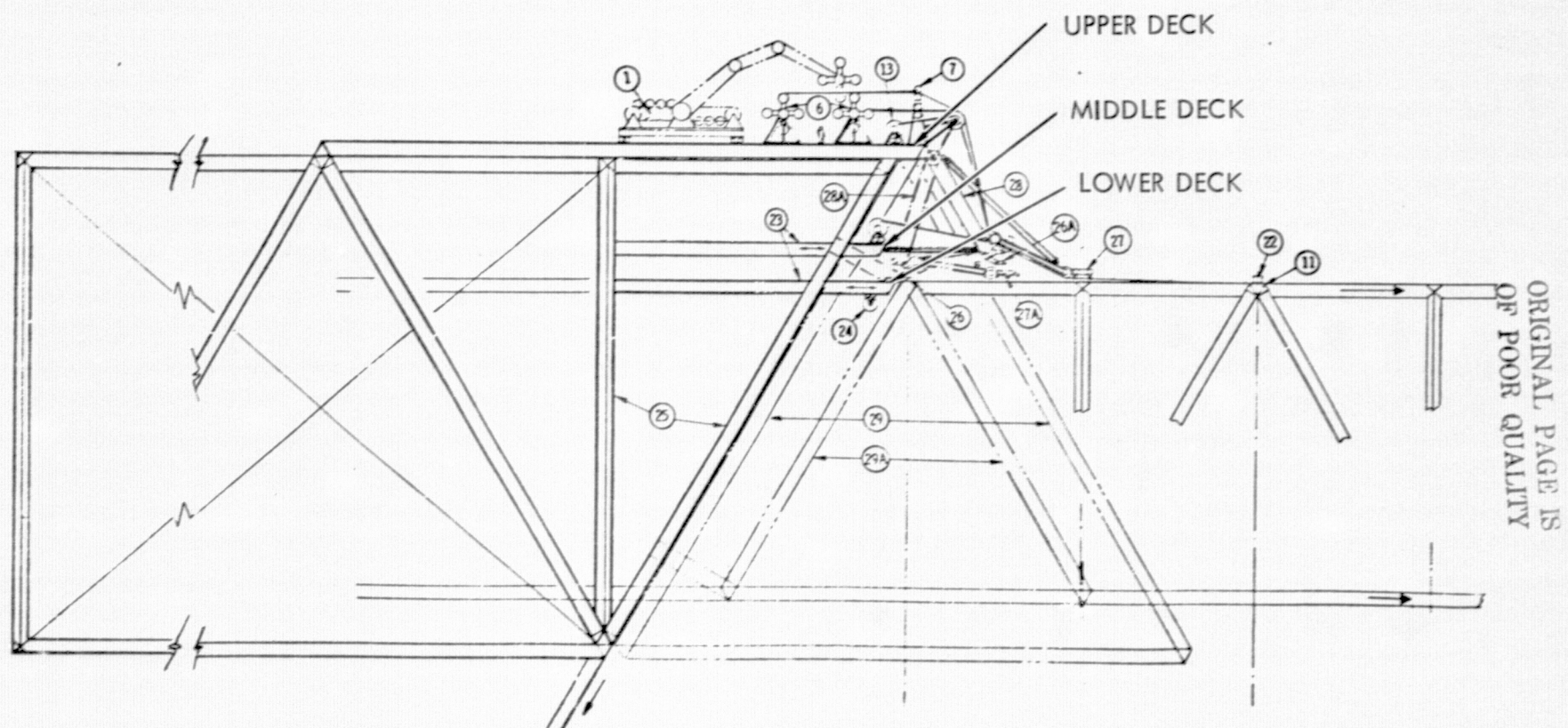
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BLANKET INSTALLATION STATION ELEVATION

This elevation shows the position of the upper, middle, and lower deck with respect to the tribeam fabricators (25), (29), together with the location of the various loading and dispensing devices. The longitudinal beam is advancing to the right as it is constructed. A portion of a completed crossbeam (11) is shown. The next crossbeam (29A) is in the fabrication position. The activity which takes place at each of the deck stations is described in more detail on the next chart.

BLANKET INSTALLATION STATION ELEVATION

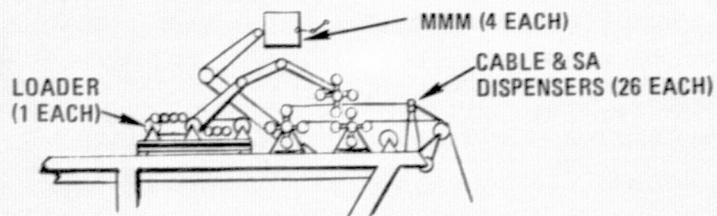


MANNED OPERATIONS AT SOLAR BLANKET INSTALLATION STATIONS

The primary operations occurring at the upper, middle, and lower deck stations during beam fabrication and solar blanket installation are identified. The locations of the manned manipulator modules (MMM) required to support the installations also are shown. These modules are mounted on transverse tracks and are spaced so that each module services approximately one-fourth of the 27 installation stations across the span of the crossbeam.

MANNED OPERATIONS AT SOLAR BLANKET INSTALLATION STATIONS

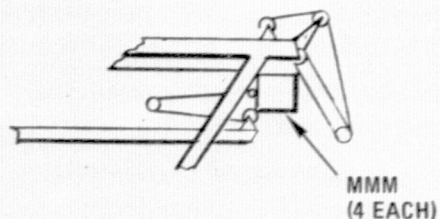
UPPER DECK



- DELIVER SA ROLLS TO LOADER
- LOADER INSERTS ROLLS IN DISPENSERS
- ROLLS THREADED THRU ROLLER SYSTEM
- DISPENSER MONITORING

4 MEN

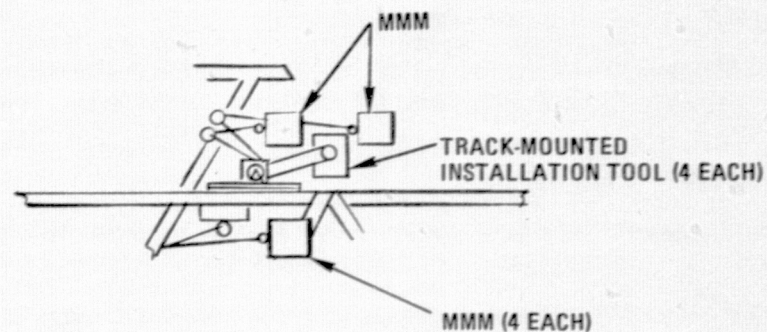
MIDDLE DECK



- FABRICATE & THREAD LONG CATENARY
- CATENARY/CABLE ATTACH MONITORING

4 MEN

LOWER DECK



- INSTALL SADDLE CLAMPS
- INSTALL SWITCH GEARS, SM & RAC ASSP
- INSTALL INSULATION MOUNTS
- ATTACH & TENSION SA'S
- INSTALL FEEDER & DM&C BUS
- MAKE ELECTRICAL CONNECTIONS

12 MEN

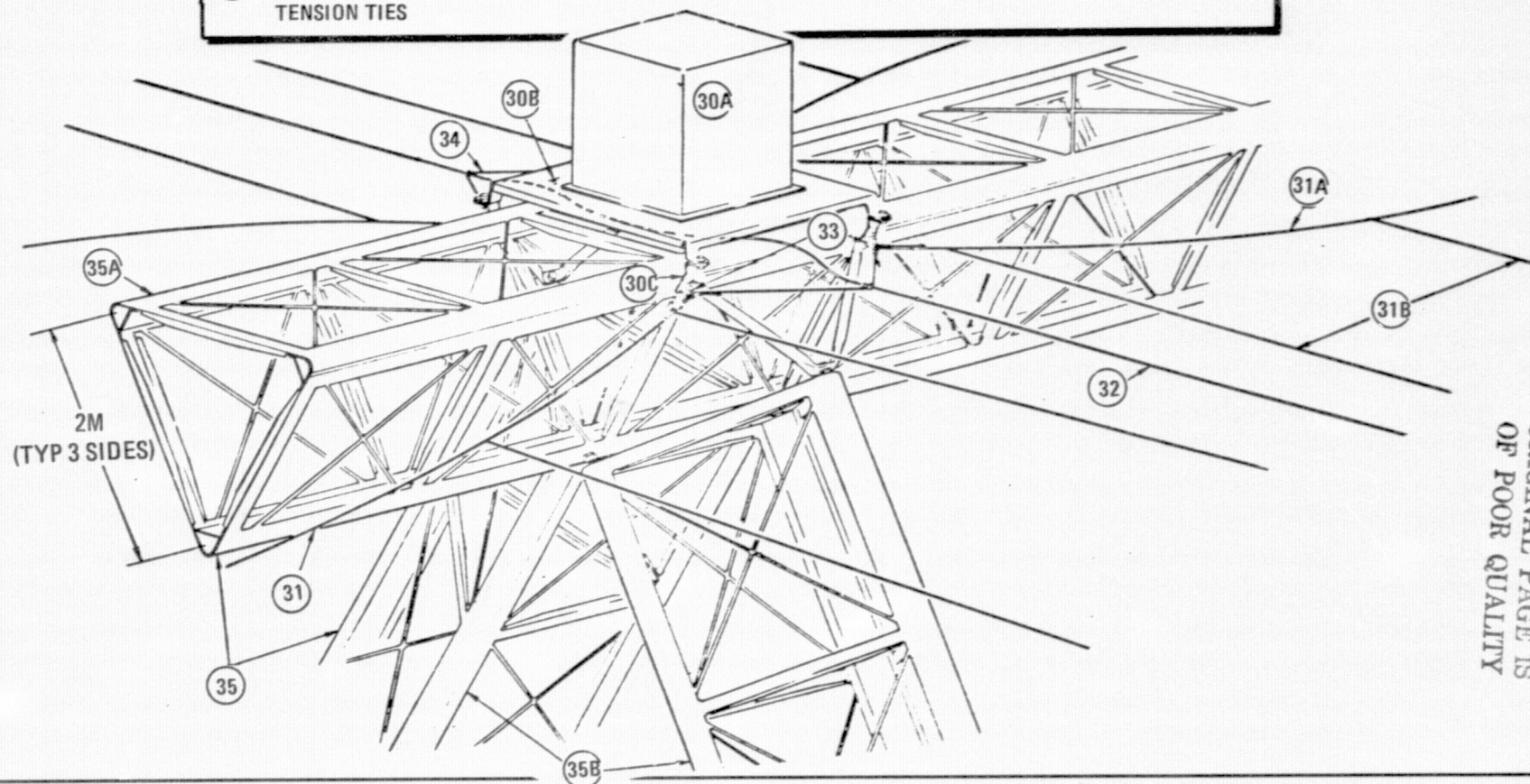
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INSTALLATION OPERATIONS AT CROSS BEAMS

Upon completion of crossbeam structure fabrication the saddle clamps, switch assemblies and DM&C equipment are the first items to be installed. Secondary feeders and their insulation mounts are included in this installation for alternate crossbeams. The SB trailing edge catenaries and longitudinal cable tensioning ties are then attached, followed by tensioning and clamping of the transverse catenaries and longitudinal cables. Following this operation, the leading edges of the SB transverse catenaries are attached to the trailing edge of the crossbeam by means of the brackets installed on the saddle. The operation at each station is completed by attaching the SB connectors to the switch assemblies, connecting the DM&C elements to the DM&C bus, and connecting the switch gears to secondary feeders (alternate crossbeams). Construction of the longerons and crossbeams for the next bay is then initiated.

INSTALLATION OPERATIONS AT CROSSBEAMS

- | | |
|---|--|
| (30) SWITCH GEAR ASSY | (32) LONGITUDINAL SOLAR BLANKET RESTRAINING CABLES |
| (30A) SWITCH GEAR | (33) CATENARY/SB CABLE ATTACH FITTINGS |
| (30B) INSTALLATION SADDLE | (34) SB CABLE TENSIONING YOKE |
| (30C) MULTI-ATTACH BRACKET | (35) CROSS BEAM AT ROUGH B
(50-M TRI-BEAM GIRDER) |
| (31) 50-M TRANSVERSE CATENARY ASSY | (35A) TOP CAP (2-M BASIC BEAM ELEMENT) |
| (31A) CATENARY | (35B) SIDE BEAM MEMBERS (50-M LONG) |
| (31B) CATENARY-TO-SOLAR-BLANKET
TENSION TIES | |



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INSTALLATION OPERATIONS AT CROSSBEAMS

Switch gear assemblies, secondary power feeders, DM&C elements sub-multiplexers and remote acquisition and control (RAC) units and DM&C buses are located on the crossbeams at the ends of the blanket strips. Alternate crossbeams mount 25 and 13 switch assemblies respectively. Saddle clamps are attached to the crossbeams at 25 m intervals to coincide with the blanket strip edges. The saddle clamp assemblies provide connectors for attachment and tensioning of cables and catenaries, mounting provisions for secondary feeder insulators, and a saddle for support of the switch assemblies.

INSTALLATION OPERATIONS AT CROSSBEAMS

0 4 8 12 16
HOURS

INSTALL & ATTACH HARDWARE

- SADDLES
- SWITCH GEAR
- SM/RAC ASSYS
- TENSION CABLES
- INSULATION MOUNTS*
- D&C BUS
- SEC. FEEDERS*

ATTACH & TENSION SOLAR ARRAY BLANKETS

- TR. EDGE OF DEPLOYED BLANKET TO CROSSBEAM LEAD. EDGE
- TR. EDGE CATENARY - TENSION & CLAMP
 - SB CABLES - TENSION & CLAMP
 - THREAD SA BLANKET
- LEAD. EDGE OF NEXT BLANKET TO CROSSBEAM TRAIL. EDGE

ELECTRICAL INSTALLATION AND CONNECTION

- SB PWR LEAD CONNECTORS TO SWITCH GEAR
- SWITCH GEAR TO SECONDARY FEEDER*
- SECONDARY FEEDER TO MAIN FEEDER*
- DM&C CONNECTORS TO SM/RAC ASSY'S
- SM/RAC ASSY'S TO D&C BUS

C/O

* ALTERNATE CROSS BEAMS

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SOLAR CONVERTER CONSTRUCTION SEQUENCE

The major operations required to complete each bay are individually timed by shift and by day. Construction of each bay is a two day process. Detailed timelines of the operations which occur at each cross-beam (i.e., CB N+1) are shown on the next chart.

SOLAR CONVERTER CONSTRUCTION SEQUENCE TYPICAL BAY

OPERATION	DAY	1			2		
	SHIFT	1	2	3	1	2	3
CONSTRUCT & INSTALL BAY N							
FAB LONGERONS & DISPENSE SB'S & REFL'S BAY N							
FAB CB N + 1							
BEAM							
END FITTINGS							
ATTACH CB N + 1 TO LONGERONS							
ALIGN STRUCTURE							
RELOAD SOLAR BLNKT & REFL. DISPENSERS							
INSTALL EQUIPMENT, MAKE ATTACHMENTS & TENSION AT CBN + 1							
INSTALL HARDWARE							
ATTACH & TENSION BLANKETS							
INSTALL/CONNECT ELECT., PWR/DM&C BUSES							
CHECKOUT							

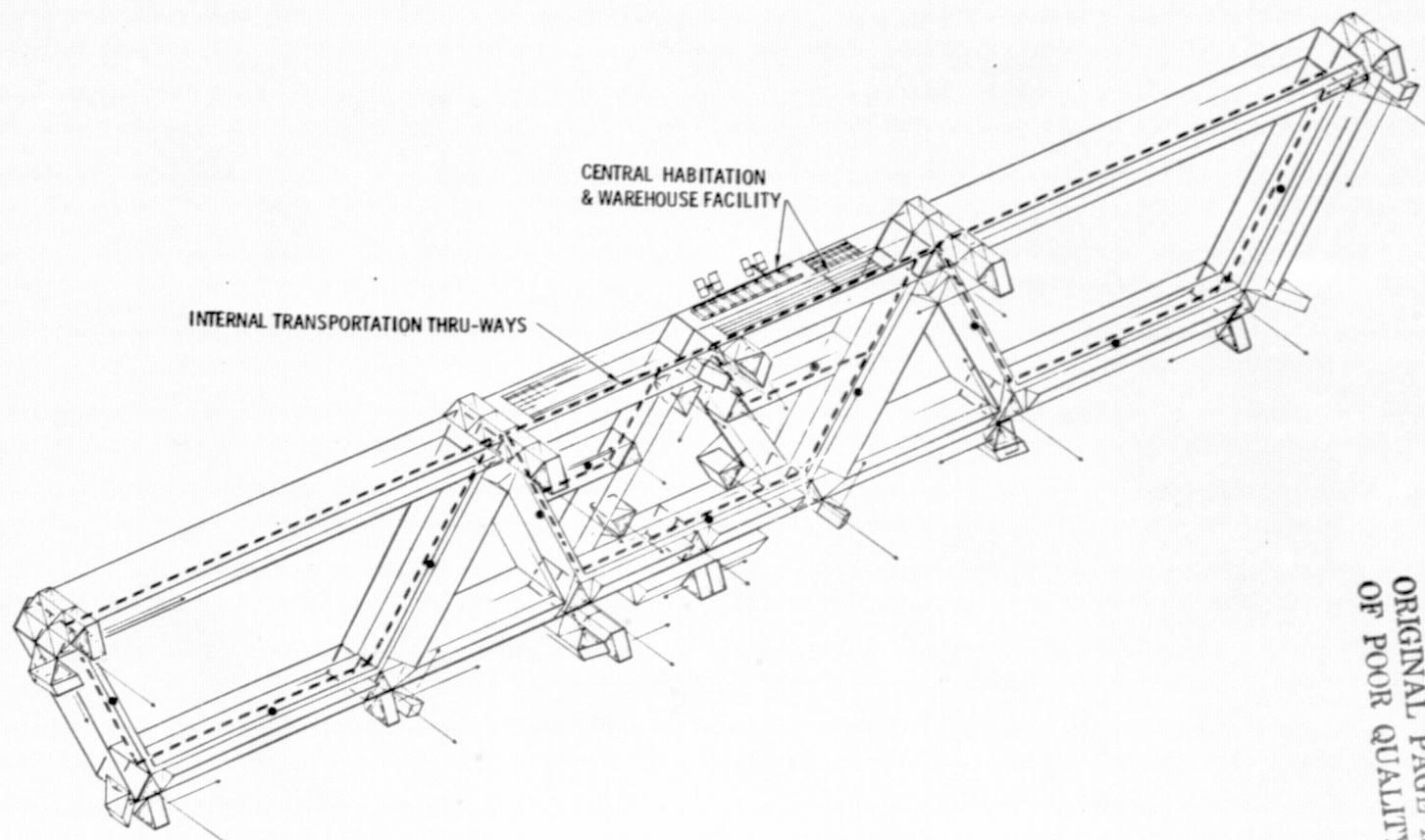
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SOLAR BLANKET TRANSPORT

Twenty-six solar blanket strips are required for each bay of each of the three troughs. A total of 78 blanket rolls must be delivered to the three locations prior to the start of the next bay construction. Three logistics vehicles capable of transporting up to 26 blanket rolls and two reflector rolls each travel on tracks between the warehouse and the three installation stations for this purpose. The chart shows the location of the tracks.

SOLAR BLANKET TRANSPORT



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SOLAR BLANKET TRANSPORT AND LOADING TIMELINE

Eight hours are required to deliver sufficient blanket rolls for one bay of installation to their dispensers. Three blanket logistics vehicles, each servicing one trough performed this task every two days.

SOLAR BLANKET TRANSPORT AND LOADING TIMELINE

● LOAD TRANSPORTER AT WAREHOUSE	ON-GOING
● TRANSPORT TO SB INSTALLATION STATION	0.7
● LOAD 26 SB DISPENSERS ($\approx$ 15 MIN./DISP.)	6.4
● TRANSPORTER RETURN TO WAREHOUSE	<u>0.9</u>
	8.0 HRS

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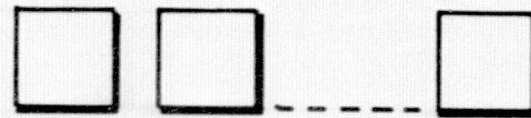


ANTENNA RF ELEMENTS PROCESSING

The processing of the RF elements at the SCB is shown schematically. By utilizing the full 180 days the entire processing can be handled by three stations, with eight men per station per shift at the very nominal rates shown. In order to utilize all the time available, the subarrays and mechanical modules for the N th + 1 satellite is initiated after completion of the antenna, and during fabrication of the 2nd wing, of the N th satellite as illustrated on a later chart which shows the overall construction schedule.

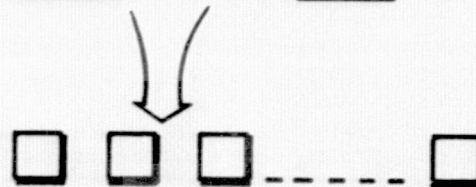
ANTENNA RF ELEMENTS PROCESSING

135684
KLYSTRONS
(120 DAYS)



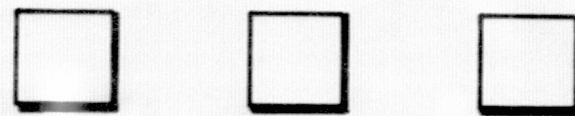
3 STATIONS: 16 KLYSTRONS/HR EACH

6993
SUBARRAYS
(120 DAYS)



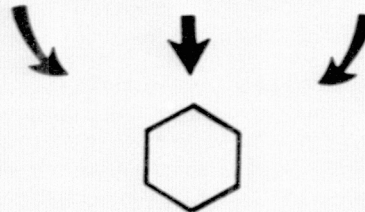
3 STATIONS: < 1 SUBARRAY/HR EACH

777
MECHANICAL MODULES
(120 DAYS ASSEMBLY
60 DAYS INSTALLATION)



3 STATIONS:
< 1 ASSEMBLY/SHIFT EACH
< 2 INSTALLATIONS/SHIFT EACH

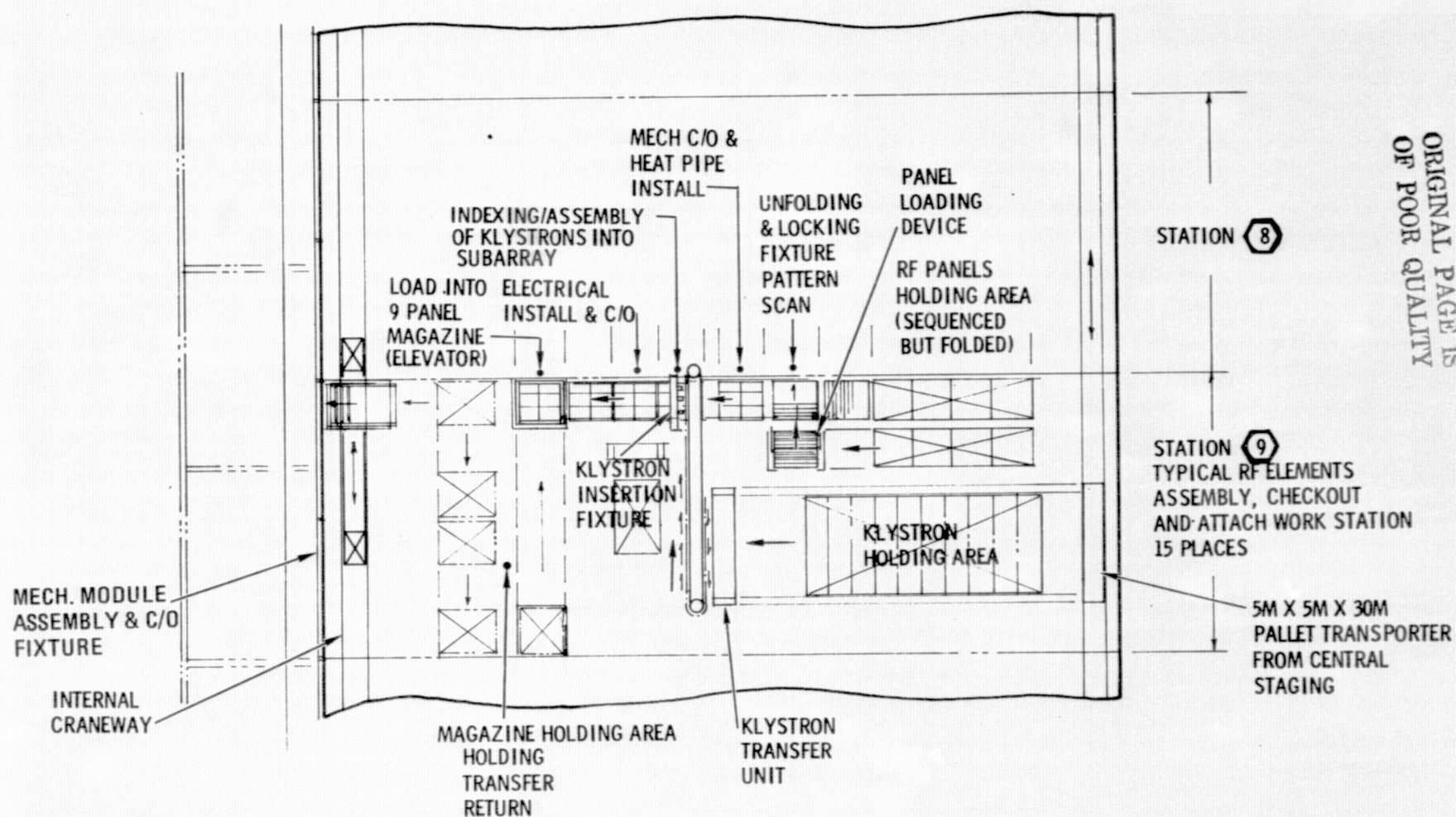
ANTENNA
(180 DAYS)



PLAN VIEW, RF ASSEMBLY STATION

At this station the waveguide subarrays are unfolded to their operational surface dimensions of approximately 11x10-m and passed through the mechanical checkout station and klystron indexing/assembly station, where the klystrons are automatically installed. The assembly of subarray and klystrons move to the electrical station where the electronic control boxes are installed, electrical connections made, and the assembly functionally checked out. Nine assemblies of identical configurations are loaded into the elevator magazine which transfers them to the mechanical module indexing/assembly crane. This tool assembles the subarrays into mechanical modules in the assembly and C/O fixture. After assembly and C/O is completed the mechanical modules are transferred to their storage location on the SCB via the mobile mechanical module transfer and installation unit. Later these same units retrieve the completed modules from the storage area, carry them out onto the installation gantry, install them on the antenna structure and checkout the completed installation.

PLAN VIEW, RF ASSEMBLY STATION (TYPICAL 3 STATIONS)

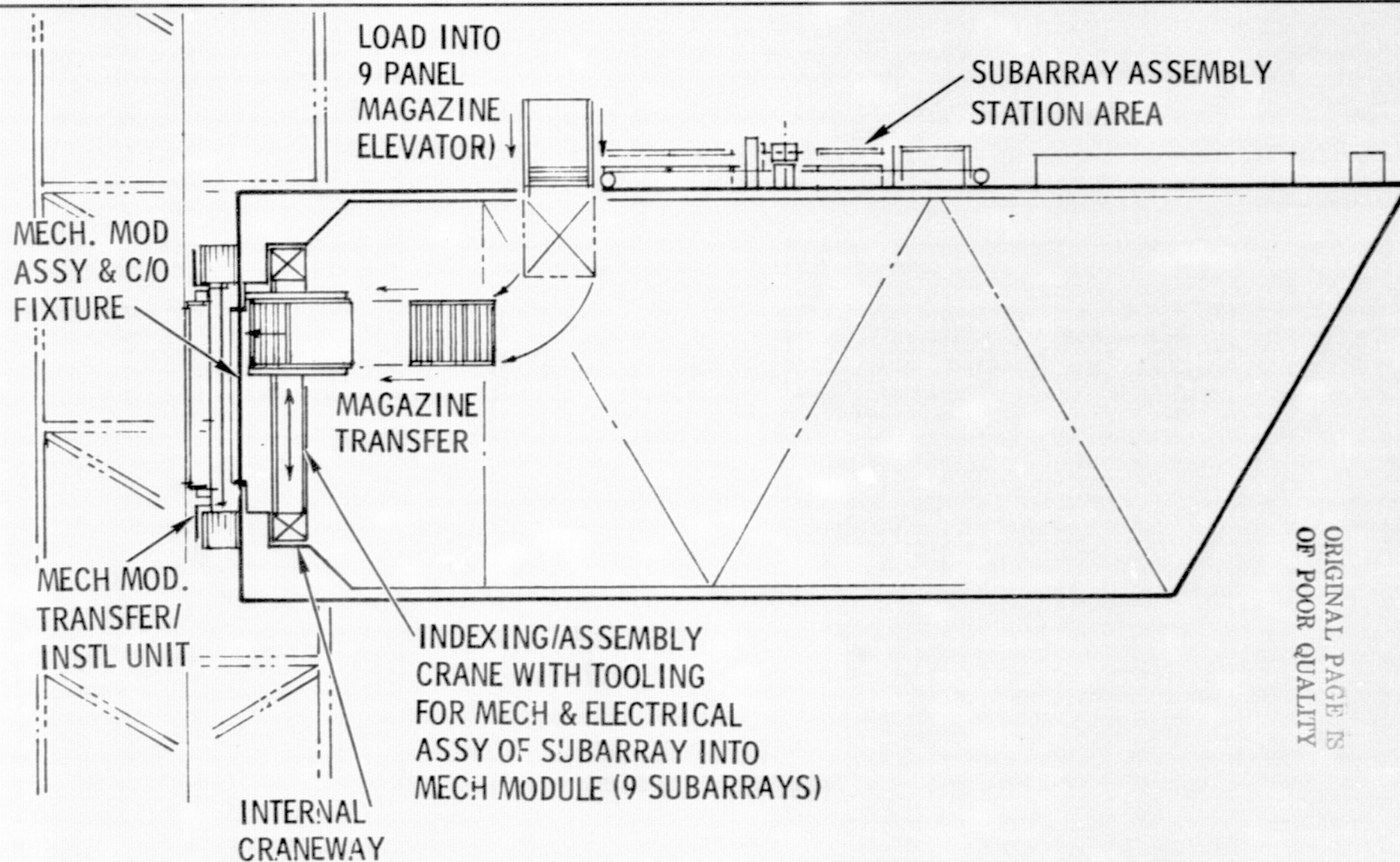


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ELEVATION, RF ASSEMBLY STATION

This elevation shows the transfer of the completed subarrays to the indexing crane and its function in assembling them into the mechanical module assy and checkout fixture.

ELEVATION, RF ASSEMBLY STATION (TYPICAL 3 STATIONS)



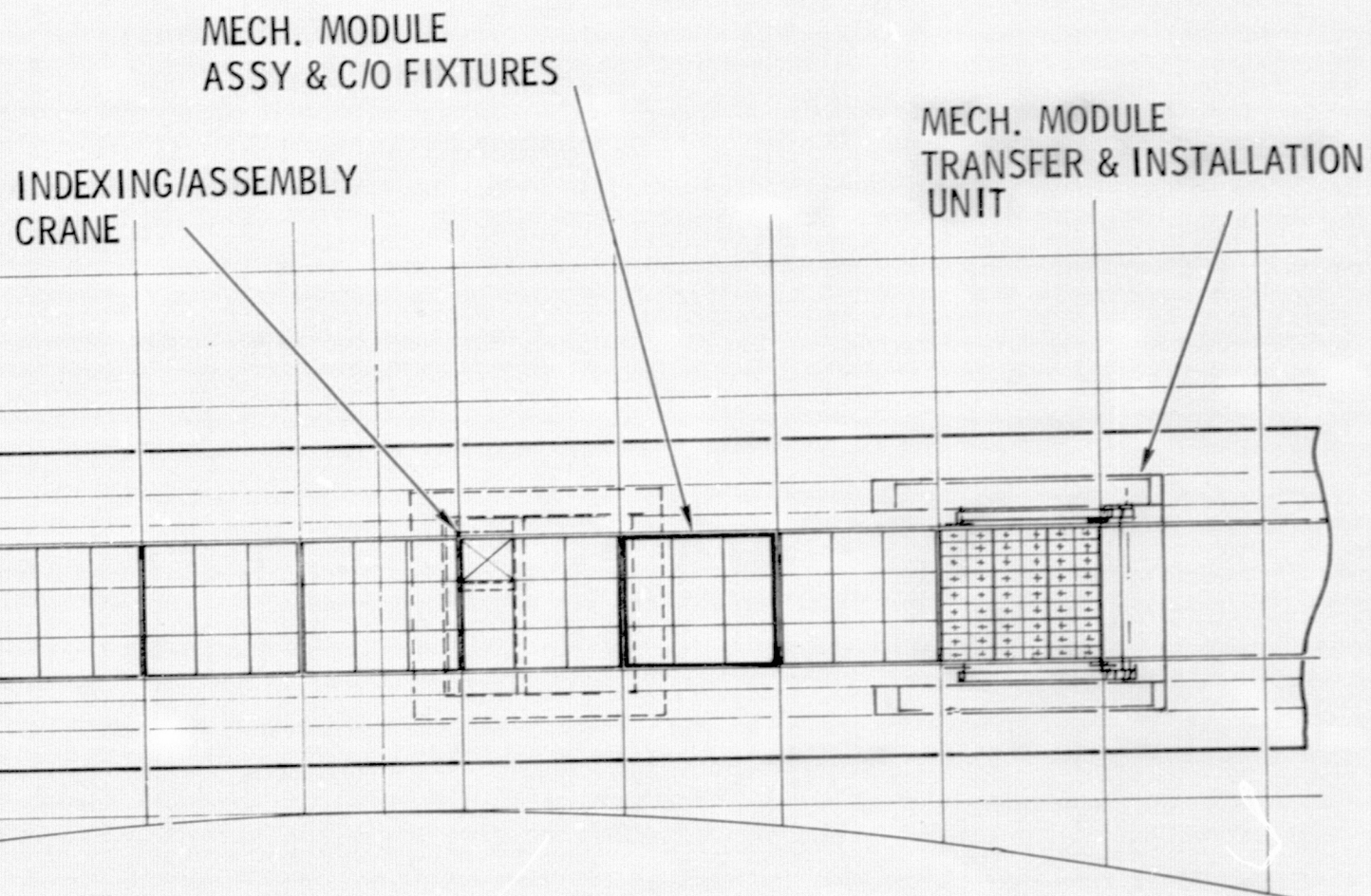
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MECHANICAL MODULE ASSEMBLY AND INSTALLATION STATION

This partial front face view shows the indexing/assembly crane, the mechanical module assembly and checkout fixture, and the mechanical module transfer and installation platform. The crane indexes the magazine to each of 9 sections of an assembly jig, installing a subarray in each, automatically making the connections and testing for mechanical module integrity. Three mobile module transfer and attach units remove the completed modules from their assembly jig and transfer them to the storage area or to the antenna installation gantry, where they are installed on the antenna structure, electrical connections made, and checked out.

MECHANICAL MODULE ASSEMBLY & INSTALLATION STATION

(PARTIAL FRONT FACE VIEW OF RF ASSY FACILITY)

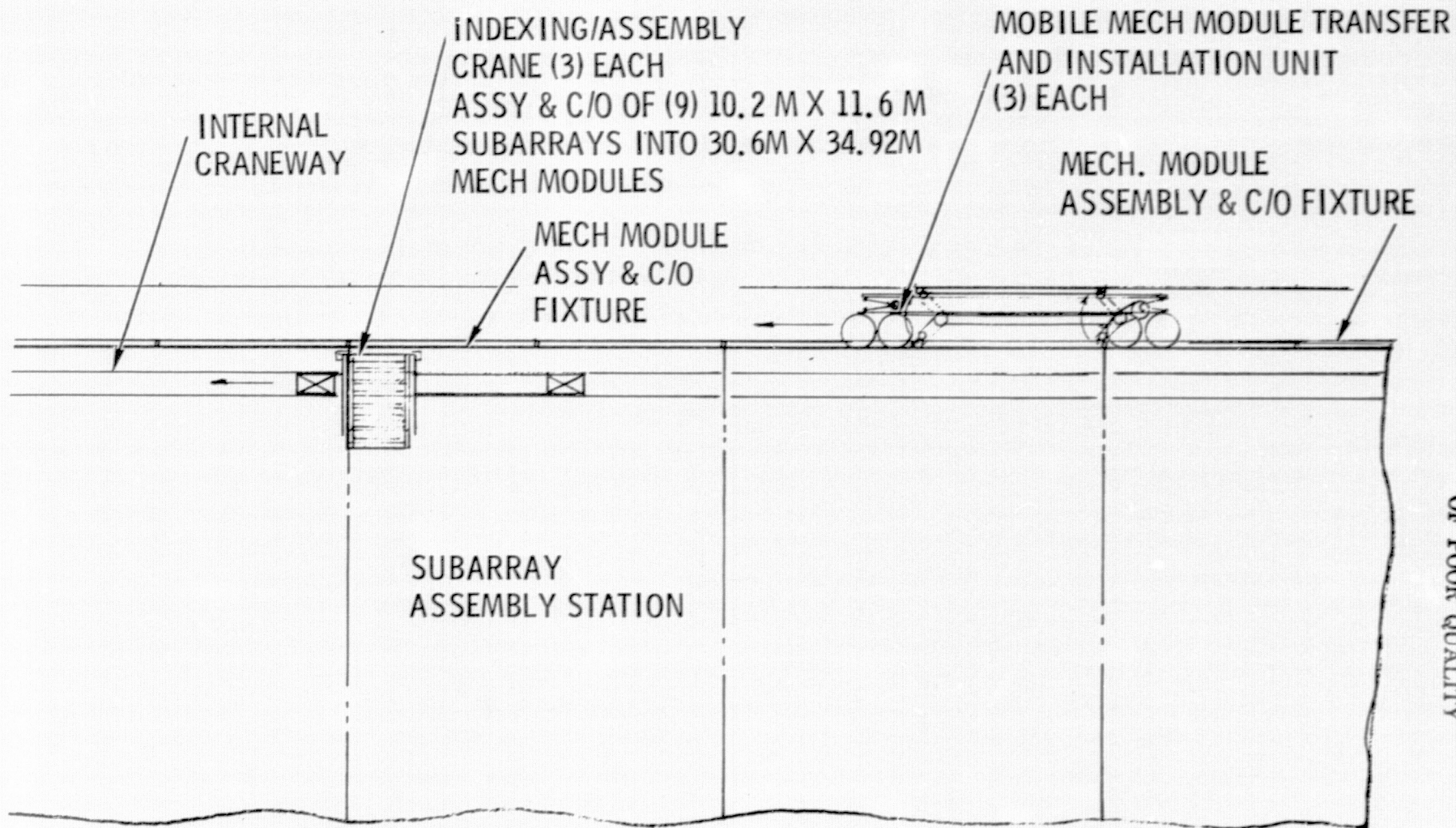


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PARTIAL PLAN VIEW OF RF ASSEMBLY STATION

This chart shows additional details of the indexing/assembly crane and the mobile mechanical module transfer and installation unit.

PARTIAL PLAN VIEW RF ASSY STATION



CREW SIZE

The time-phased crew requirements for the construction concept and 180 day construction schedule described are summarized here.

CREW SIZE
(3-TROUGH, CENTER ANTENNA, SINGLE PASS)

	DAY 1-90	DAY 91-150	DAY 151-180
<u>CONSTRUCTION CREW</u>			
SOLAR CONVERTER CONSTRUCTION	60	0	0
ANTENNA RF ASSY/INSTALLATION	24	24	24
CONSTRUCTION SUPPORT	12	4	4
MAINTENANCE	10	3	3
BASE MGMT	8	3	3
CREW SUPPORT	<u>12</u>	<u>4</u>	<u>4</u>
TOTAL SHIFT CREW	126	38	38
TOTAL CONSTN. CREW - 4 SHIFTS	504	152	152
<u>OPERATIONAL MAINTENANCE CREW</u>			<u>24</u>
TOTAL CREW ON-BOARD	504	152	176

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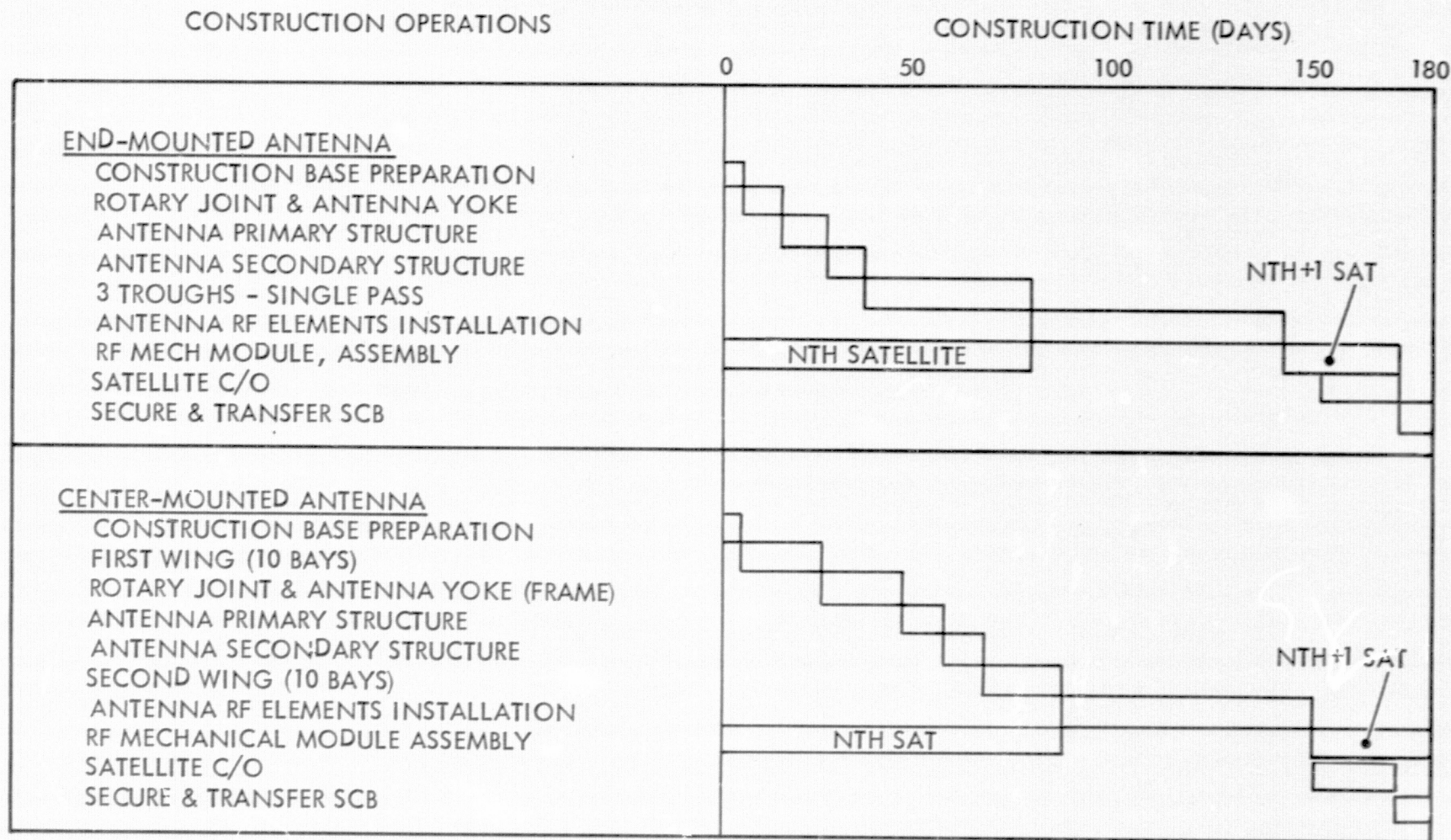


SATELLITE CONSTRUCTION TIMELINE COMPARISON
(3 TROUGH SATELLITE, SINGLE PASS CONSTRUCTION)

A timeline has been developed for construction of a 3-trough satellite with an end mounted antenna using the single pass construction techniques. For comparison purposes this schedule is shown along with the (previously presented) schedule for the same satellite with a center mounted antenna. Evaluation of the constructability of both antenna locations has shown that the end mount is somewhat simpler thereby requiring only 10 days to construct vs 20 days for the center mount. This will result in a slightly reduced manpower requirement (<6.0%) which can be reflected in a somewhat reduced crew size or a shortened tour of duty for the construction crew which starts on day 91. Otherwise the construction schedule is effectively independent of antenna location.

SATELLITE CONSTRUCTION TIMELINE COMPARISON

(3 TROUGH SATELLITE, SINGLE PASS CONSTRUCTION)



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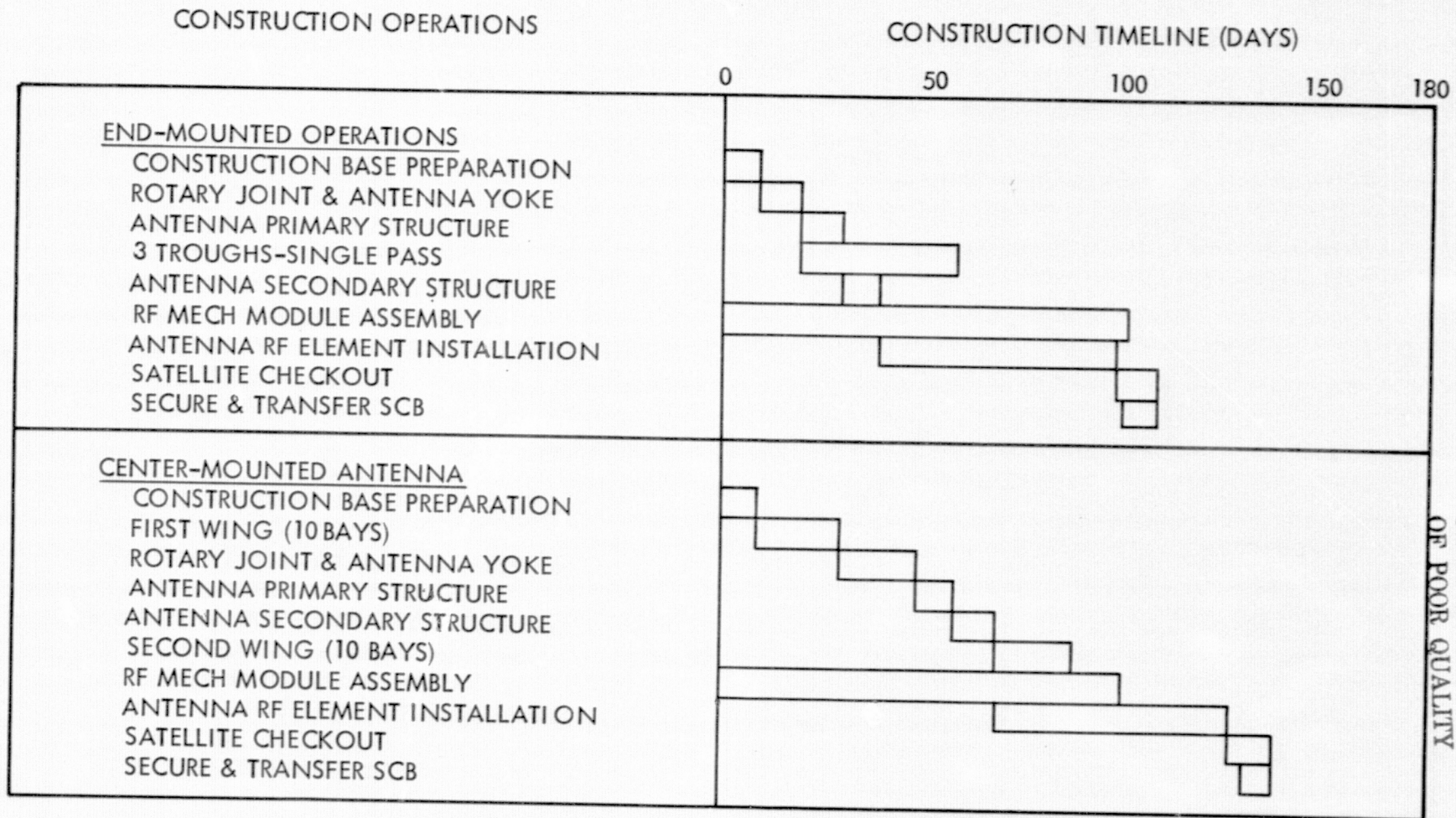


SATELLITE SINGLE PASS CONSTRUCTION TIMELINE (ACCELERATED SCHEDULE)

The nature of the single pass construction concept makes possible the shortening of the 180 day schedule through the techniques described below. Such shortening; however, requires that certain operations be conducted in parallel, with a resultant increase in crew size. For example, the completion schedule of 108 days has been achieved here for the end-mounted and 127 days for the center-mounted antennas by increasing the (average) crew sizes from 332 to 376 and to 438 for the two cases respectively.

Timelines for both the center-mounted and end-mounted antennas are shown. Since the single pass concept constructs all three troughs simultaneously as opposed to the sequential serpentine operation, the total construction time can be shortened. As with the 4 trough configuration, considerable time for mechanical module assembly is allocated and parallels, in part, antenna RF installation. In the case of the end-mounted antenna, the rotary joint and antenna yoke are constructed initially. Antenna construction then progresses in parallel with trough construction. The center-mounted antenna requires construction of one wing before the rotary joint and yoke can be fabricated and installed. The time for this operation has been doubled over the end-mounted configuration to reflect the additional operational complexities of installation. After the antenna yoke has been installed, the second satellite wing is fabricated, but not as a pacing item because of the time required for antenna RF installation.

SATELLITE SINGLE PASS CONSTRUCTION SCHEDULE (3 TROUGHS, SPACE FRAME ANTENNA)



ANTENNA CONSTRUCTION CONCEPT



Rockwell International
Space Division

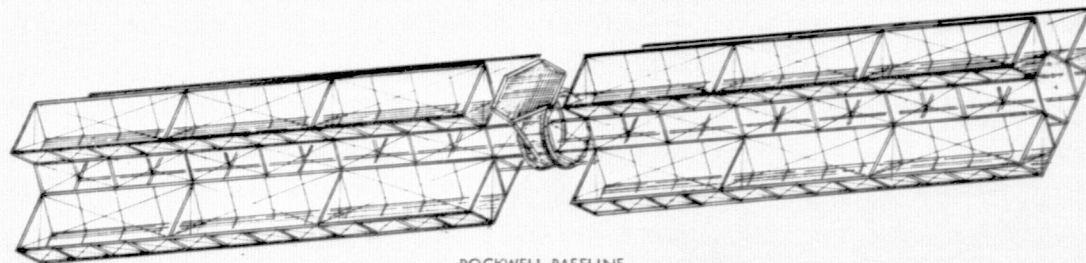
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300
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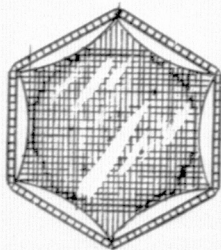
ANTENNA DESIGN CONCEPTS

End and center-mounted three trough satellite configurations are shown for both the tension web and space frame antenna. While the space frame antenna has been developed as the preferred concept, the alternate tension web version also is shown inasmuch as the configuration is considered to have some merit, particularly with regard to relative ease of construction and certain thermal advantages.

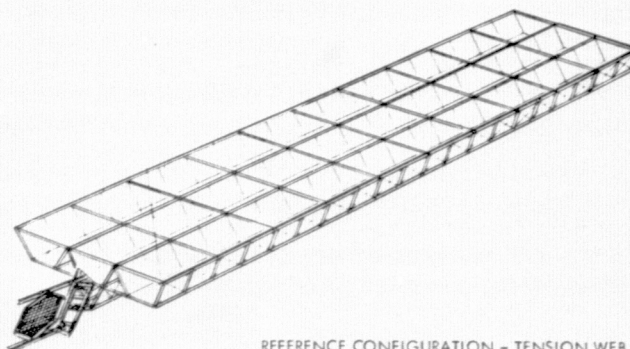
ANTENNA DESIGN CONCEPTS



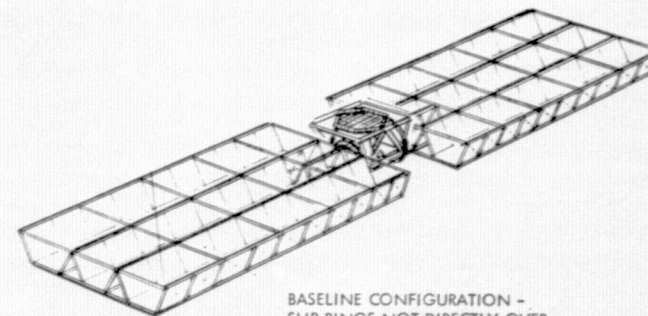
ROCKWELL BASELINE



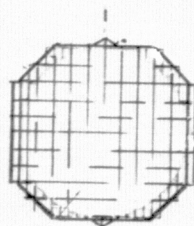
TENSION WEB



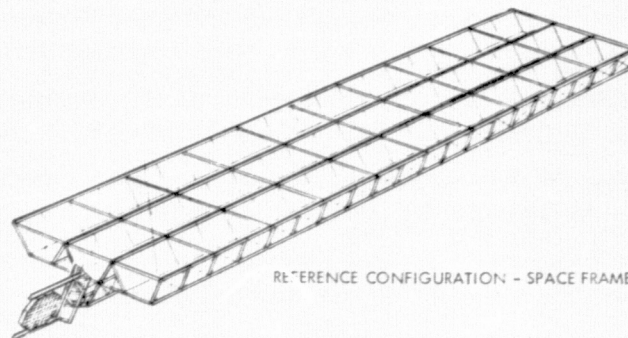
REFERENCE CONFIGURATION - TENSION WEB



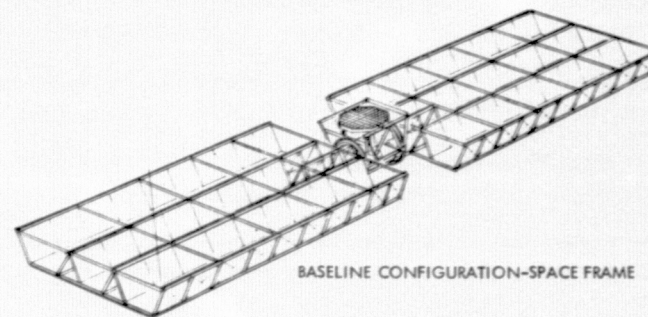
BASELINE CONFIGURATION -
SLIP RINGS NOT DIRECTLY OVER
ANTENNA



SPACEFRAME



REFERENCE CONFIGURATION - SPACE FRAME



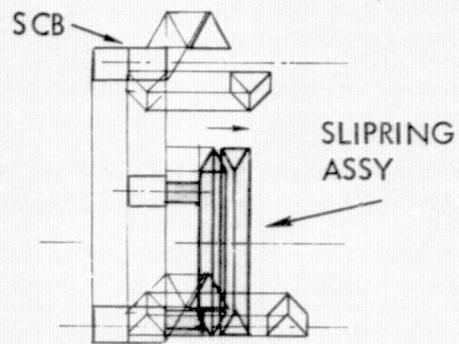
BASELINE CONFIGURATION-SPACE FRAME

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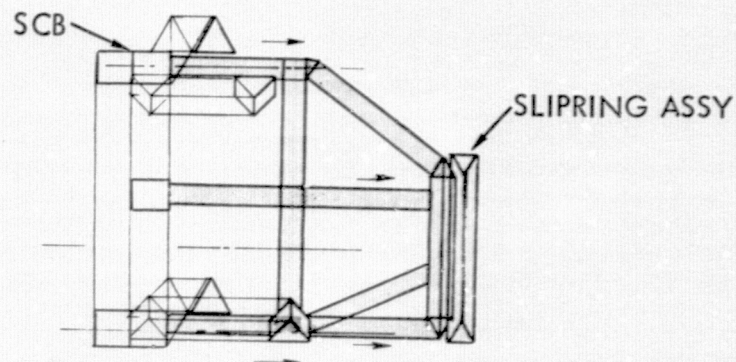
ROTARY JOINT CONSTRUCTION SEQUENCE FOR END MOUNTED ANTENNA

The sequence of fabricating the rotary joint structure during the single pass satellite construction concept is shown. The primary operations which take place are identified under each figure.

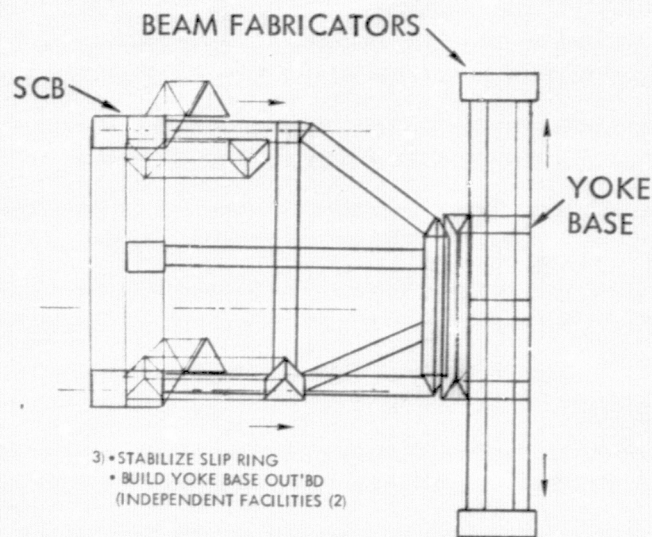
The slip ring assembly is the first element of the satellite to be constructed. It is fabricated on a jig which is part of the SCB. When the slip ring is complete the SCB fabricates the longerons which support the slip ring as the SCB moves away from it. The first frame of the solar converter is then constructed by the SCB and the slip ring is further established by apex support members which also tie the ring to the first frame. The construction of the antenna yoke is described in some detail in a later section of this document.



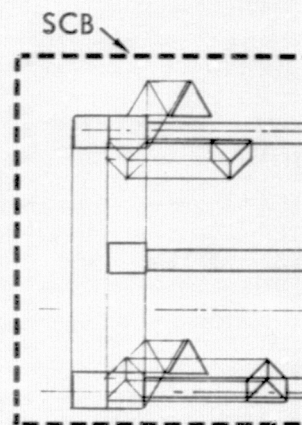
- 1) • SINGLE PASS FABR. FACILITY
- SLIP RING ASSY BUILT ON JIG
- ATTACH LONG'NS 3 PLACES



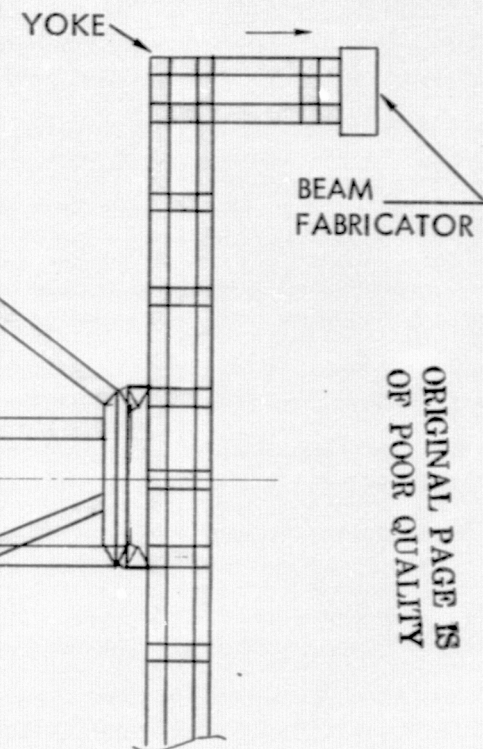
- 2) • SLIP RING ASSY EXTENDED VIA LONG'NS
- FIRST FRAME BUILT STA. O
- ATTACH APEX SUPPORT MEMBERS TO S.R. ASSY



- 3) • STABILIZE SLIP RING
- BUILD YOKE BASE OUT'BD
- (INDEPENDENT FACILITIES (2))



- 4) • BUILD YOKE ARMS

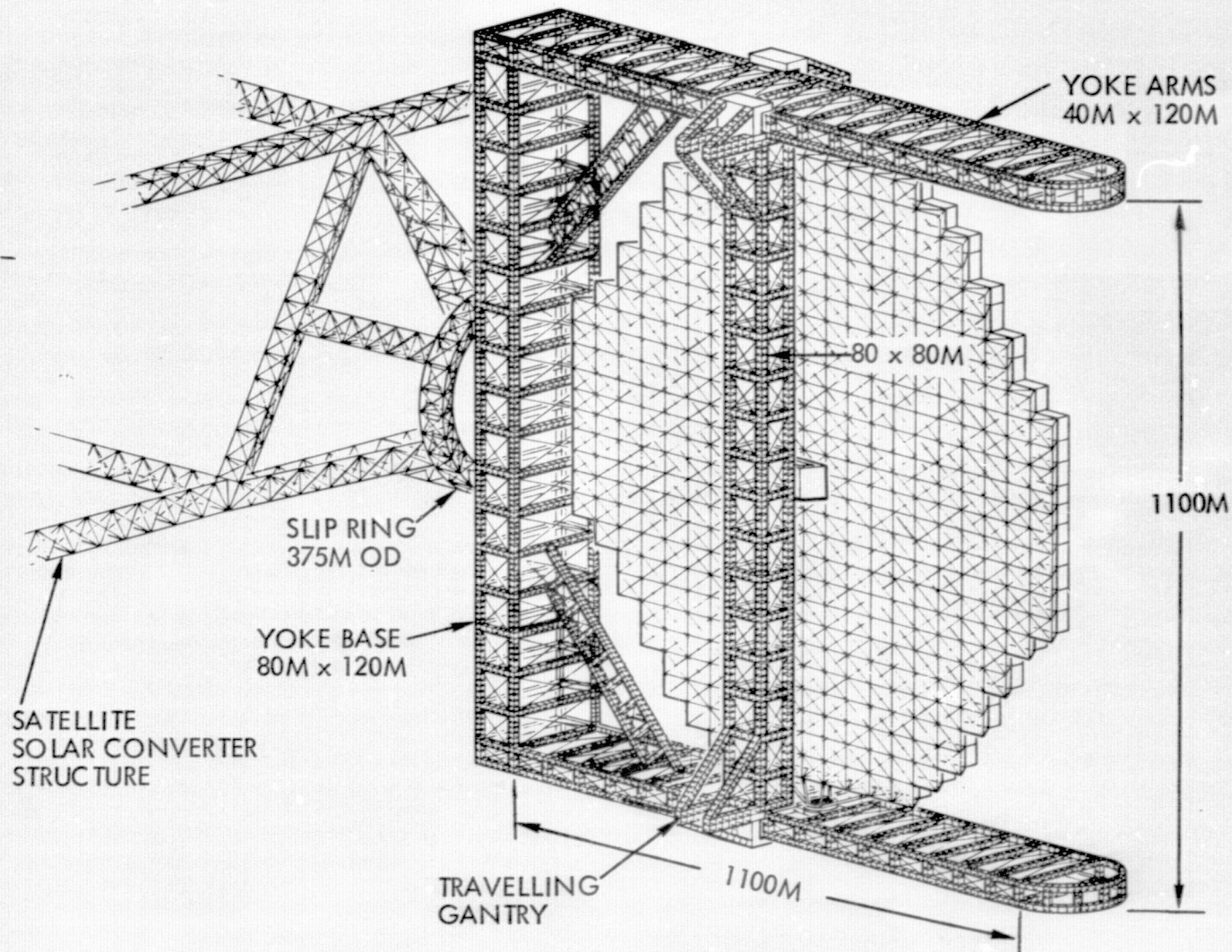


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SPACE-FRAME ANTENNA CONSTRUCTION CONCEPT END MOUNTED

The space frame antenna consists of a primary beam structure upon which a secondary structure which supports the RF elements is attached. A completed antenna structure is shown mounted in its yoke arms which, in turn, are attached to the yoke base. The rotary joint which supports the yoke base is attached to the end of the satellite, not shown in its entirety. A traveling gantry, mounted on the yoke arms, provides support for the fabricators which constructs the antenna primary structure and later provide access to the antenna for installing the RF elements. The gantry can traverse the end of the trunnion arms, thus providing access to both sides of the antenna for subsequent maintenance. When not in use it is stored adjacent to the trunnion base to avoid interference with the microwaves.

SPACE-FRAME ANTENNA CONSTRUCTION CONCEPT END MOUNTED



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ANTENNA PRIMARY STRUCTURE FABRICATION

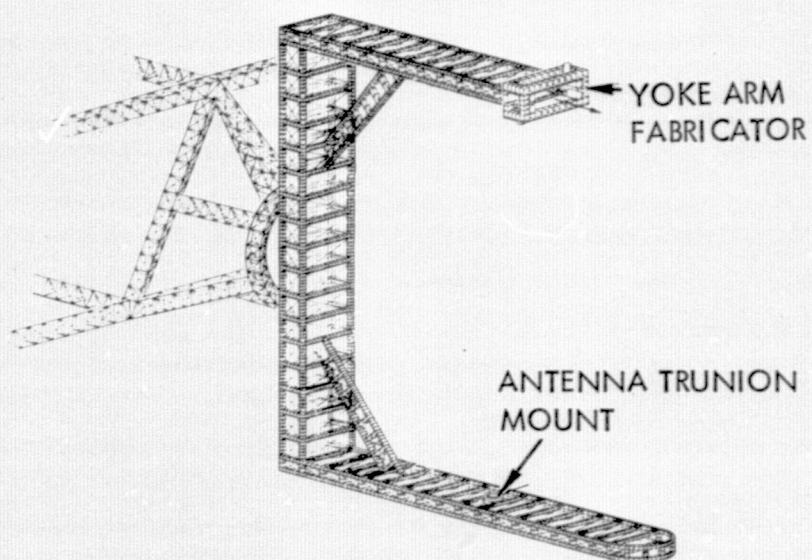
This series depicts the sequence of antenna construction operations. Initially, the yoke base is constructed in place across the face of the rotary joint utilizing a beam fabricator (or two beam fabricators working in opposite directions) which is free flown from its storage location on the SCB into its initial position and attached to the slip ring structure.

Upon completion of the yoke base, the beam fabricator is repositioned to construct each yoke arm as shown. The strengthening ties at the corners are fabricated elsewhere on the facility and then moved into place.

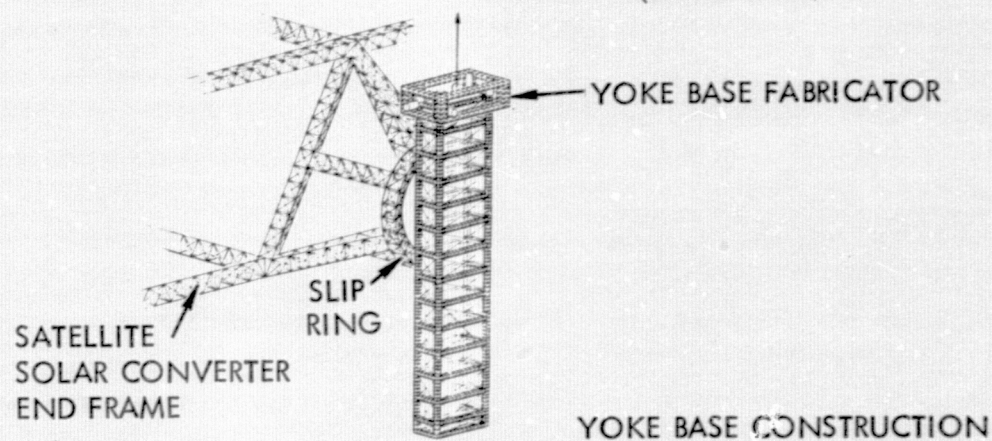
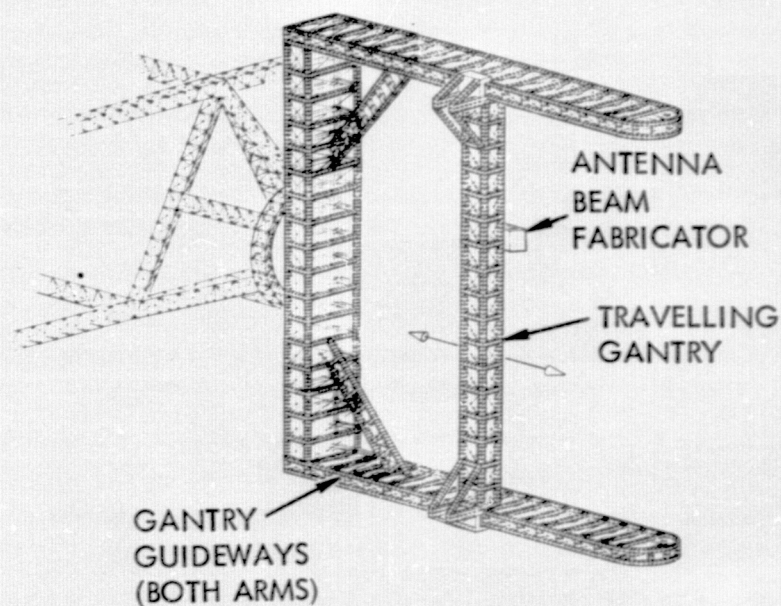
Following completion of the yoke arms, a beam fabricator is used to construct utilizing the yoke base as a platform. The gantry is then attached to tracks on the yoke arms. Elevating mechanisms at each end of the gantry provide for moving it to greater or lesser depths within the arms as may be required in the antenna construction and RF mechanical module installation operations. The elevating mechanisms also provide for raising the gantry clear of the structure for stowing along the yoke base when not in use.

ANTENNA PRIMARY STRUCTURE FABRICATION

YOKE ARM CONSTRUCTION



YOKE AND GANTRY INSTALLATION

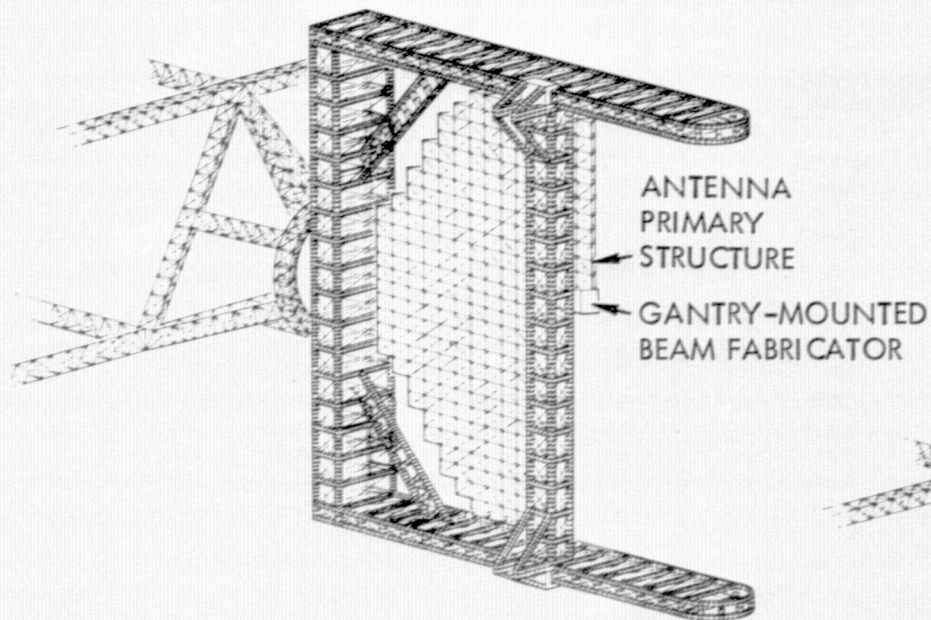


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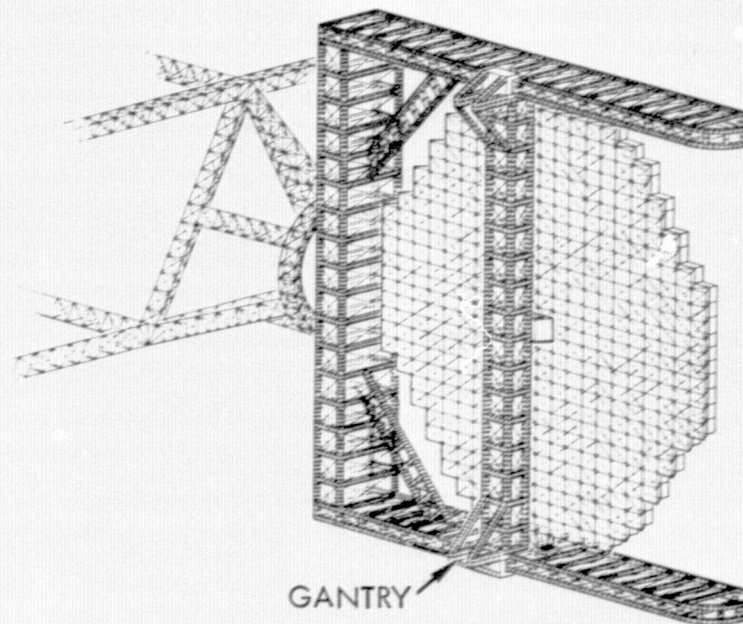
ANTENNA PRIMARY STRUCTURE FABRICATION

The antenna primary structure is constructed by beam fabricators mounted to the lower side of the gantry. Initially, the antenna center beam structure which attaches to the trunnions is fabricated and installed in the trunnions, which are then locked into position. Following this operation, the gantry-mounted fabricator progress outward from the center beam, completing one-half of the structure, in successive passes. The gantry is then relocated and the fabricator constructs the remaining half of the antenna. After removal of the fabricator, the gantry is then used for installation of secondary structure and RF elements.

ANTENNA PRIMARY STRUCTURE FABRICATION



ANTENNA PRIMARY STRUCTURE
FABRICATION



SPACE-FRAME ANTENNA CONSTRUCTION CONCEPT
END MOUNTED

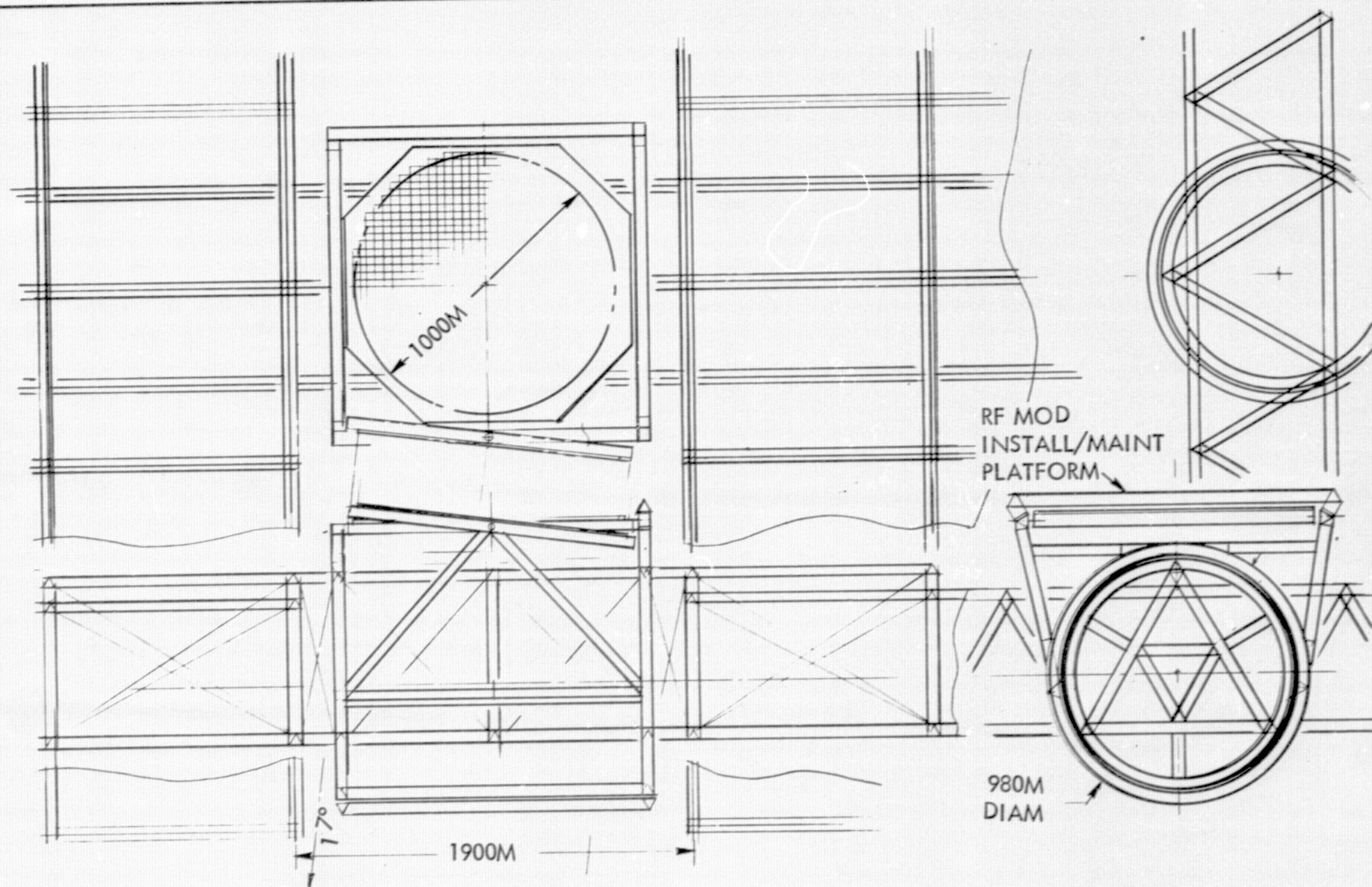
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3 TROUGH CENTER MOUNTED ANTENNA CONCEPT

This chart shows more details of a center-mounted antenna for the 3 trough satellite configuration. The slip rings are installed around longerons which continue through the center section and provide the structural tie between the two solar converter wings. The antenna is mounted within a frame which provides support for the trunnions and also for the traveling gantry used to construct the satellite structure and install the RF elements. Because of thermal considerations inherent in antenna operations the slip rings are located outside the projection of the antenna.

3 TROUGH CENTER MOUNTED ANTENNA CONCEPT

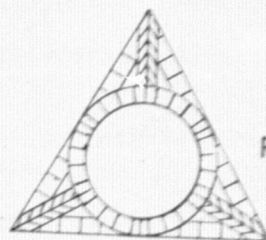
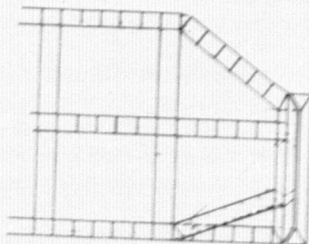
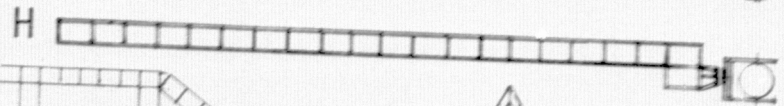
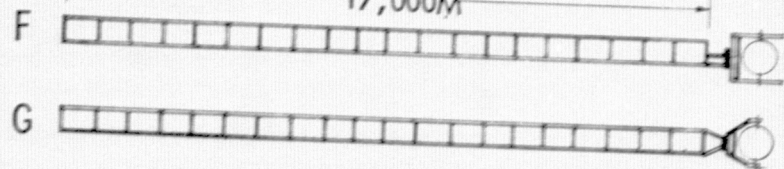
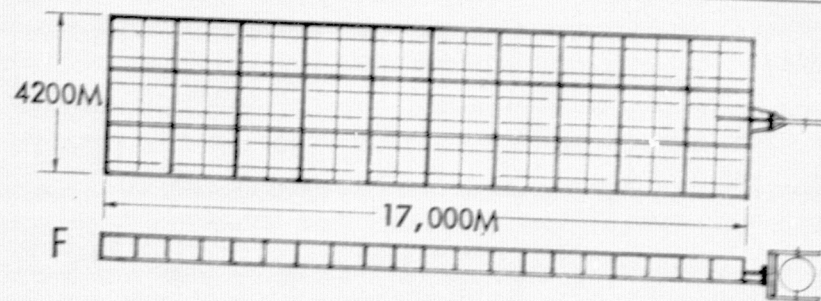
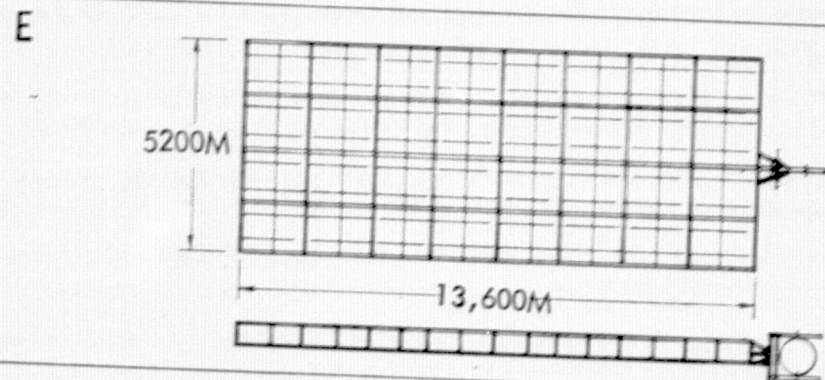


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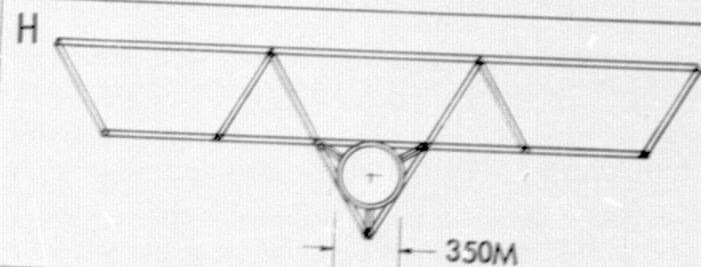
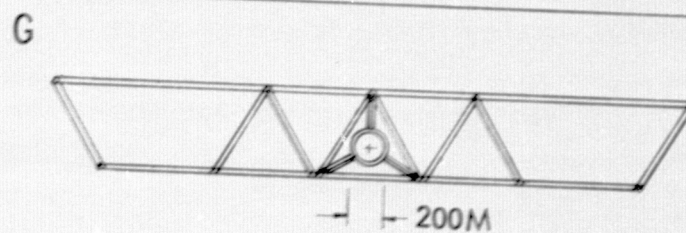
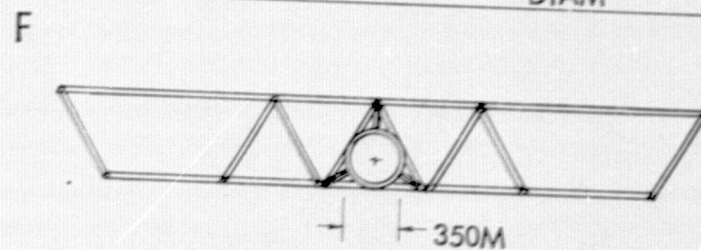
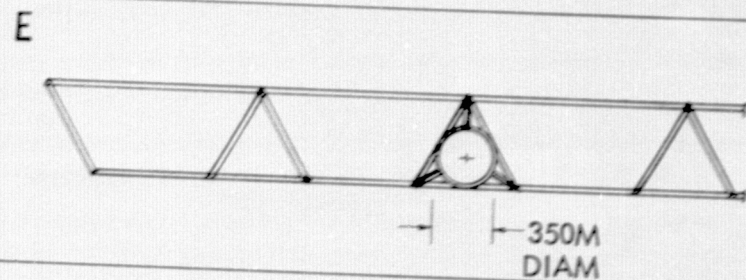
SLIP RING CONFIGURATIONS END MOUNTED ANTENNA

Potential locations of the end-mounted antenna relative to the satellite frame varies with the number of troughs as illustrated. The three trough version provides greater latitude for mounting in a manner compatible with the basic frame structure. Although not illustrated, larger size rotary joints similar to those shown for the center mounted can also be installed on the end. The downward size limitation on the joint is contingent on a stress analysis to determine the required strength of the supporting structural members.

SLIPRING CONFIGURATIONS END MOUNTED ANTENNA



END MOUNTED
ROTARY JOINT
STRUCTURAL
CONCEPT



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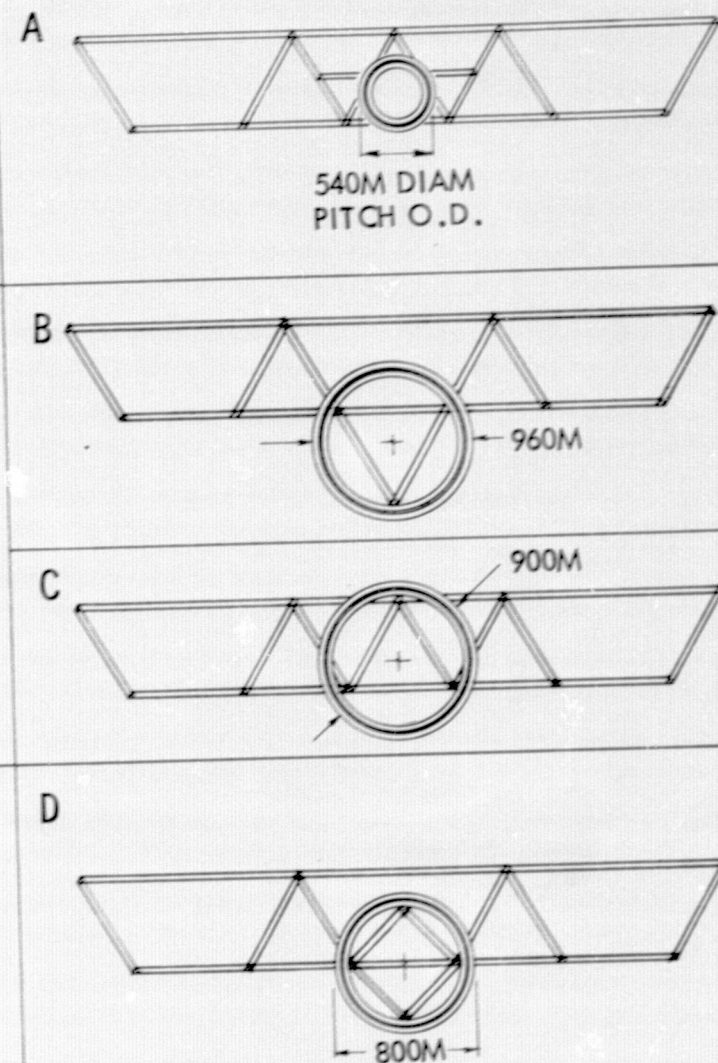
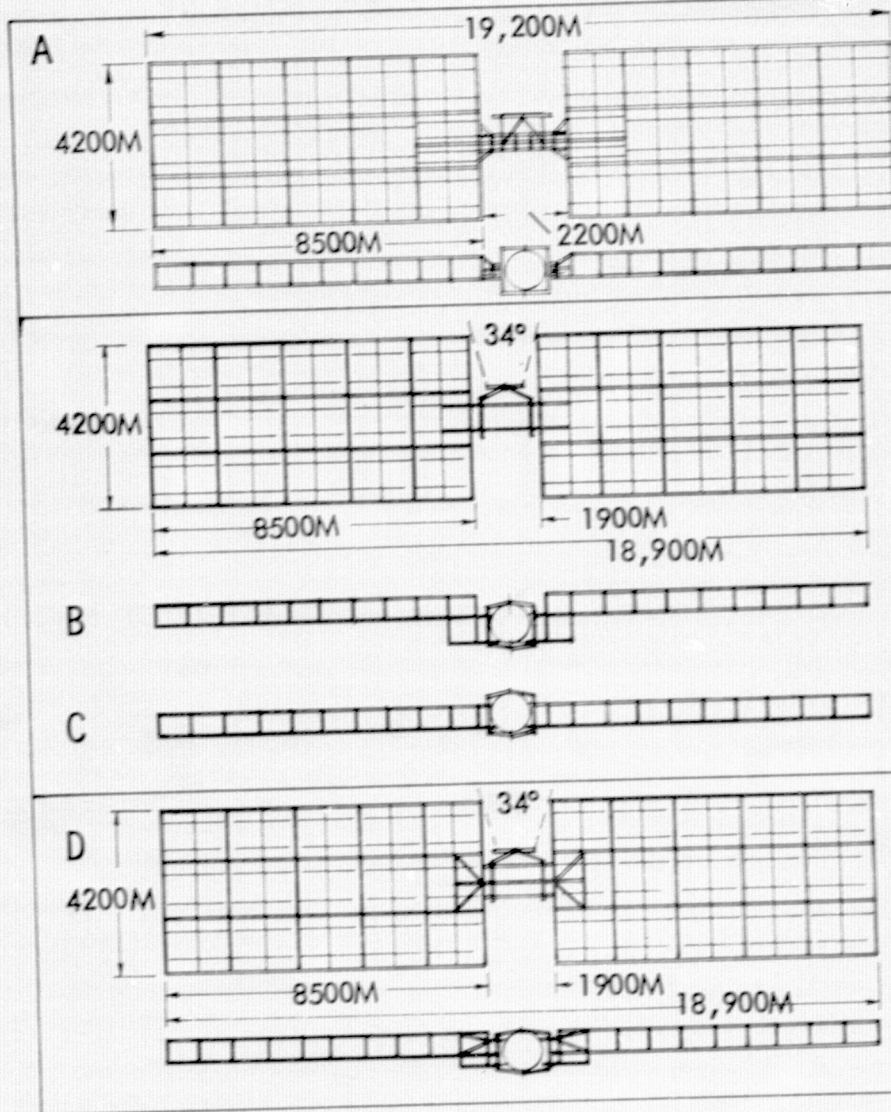


SLIP RING CONFIGURATIONS CENTER MOUNTED ANTENNA

Center-mounted antenna concepts for a three trough satellite are shown. Since the antenna can be rotated in the trunnions 17° to either side of its nominal position, the space between satellite wings must be sufficient to prevent interference by the structure with the microwaves - i.e., accommodate a 34° angle as shown.

A primary design consideration for the center-mounted antenna is the diameter of the rotary joint. A small joint is more desirable from the constructability, thermal, and weight aspects; however, this must be weighed against the ability of the supporting structure to withstand torsional and bending moments between the two wings.

SLIPRING CONFIGURATIONS CENTER MOUNTED ANTENNA



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CONSTRUCTION COMPARISONS AND CONCLUSIONS



CONSTRUCTION COMPARISON

The differences in facility mass, crew sizes, construction equipment, and construction complexity for the serpentine and single pass construction concepts are shown. The satellites evaluated consist of the three and four trough configurations with either end-mounted or center-mounted antenna. (The effect of this variation on the construction time, crew size and supporting equipment is negligible.) The serpentine concept requires a pass for each trough as opposed to the single pass concept wherein all troughs are constructed simultaneously. The relative complexity (or program risk) considers the operations attendant to fixture and platform translation required for serpentine construction, as opposed to the single pass concept.

The crew sizes reflect average manloading, since the sequence of construction operations, particularly with the single pass concept, permits returning of some personnel to earth prior to satellite completion.

The support equipment requirements (e.g., tribeam fabricators) vary with the construction concept in that for a single pass construction, all troughs and solar converter equipment are completed simultaneously instead of in series. However, the serpentine fixture is required to operate from both sides, which requires two sets of dispensing equipment.

It is noted that the serpentine method results in a smaller crew sizes, and in general, less supporting equipment. The SCB mass for the two concepts is essentially the same, with the platform accounting for a large percentage of the serpentine SCB mass. However, the precursor operations attendant to constructing a platform almost 3 km long in three sections which translate relative to one another are formidable. Moreover, the sequence of translating these sections and the construction fixture many times during the construction of one satellite involves considerable operational complexity and risk. In addition, the concept involves several sequences of securing and releasing the platform to and from the partially completed satellite structure (2 meter tribeam sections) by means of elevating attach mechanisms. Detailed study will be required to evaluate the feasibility of this operation relative to the stress concentrations involved. For these reasons, the single pass concept is preferred.

CONSTRUCTABILITY COMPARISON (180 DAY CONSTRUCTION SCHEDULE)

CONSTRUCTION CONCEPT	SATELLITE CONFIGURATION	PASSES	CONST. COMPLEX	CREW SIZE (AVG)	SCB MASS KG $\times 10^{-6}$ *		BEAM FABRI- CATORS		FREE- FLYERS		MANNED MANIP. MODULES CREW/CARGO LV'S
							SOLAR CONV.	ANTENNA	SOLAR CONV.	ANTENNA	
MULTI-PASS SERPENTINE	FOUR TROUGHS	4	6	266	5.2	22	4	4	6	16	5
	THREE TROUGHS	3	4.5	266	5.2	22	4	4	6	16	5
SINGLE PASS	FOUR TROUGHS	1	1.33	392	6.5	32	4	8	6	64	20
	THREE TROUGHS	1	1	332	5.3	24	4	6	6	48	15

*INCLUDES 25% GROWTH

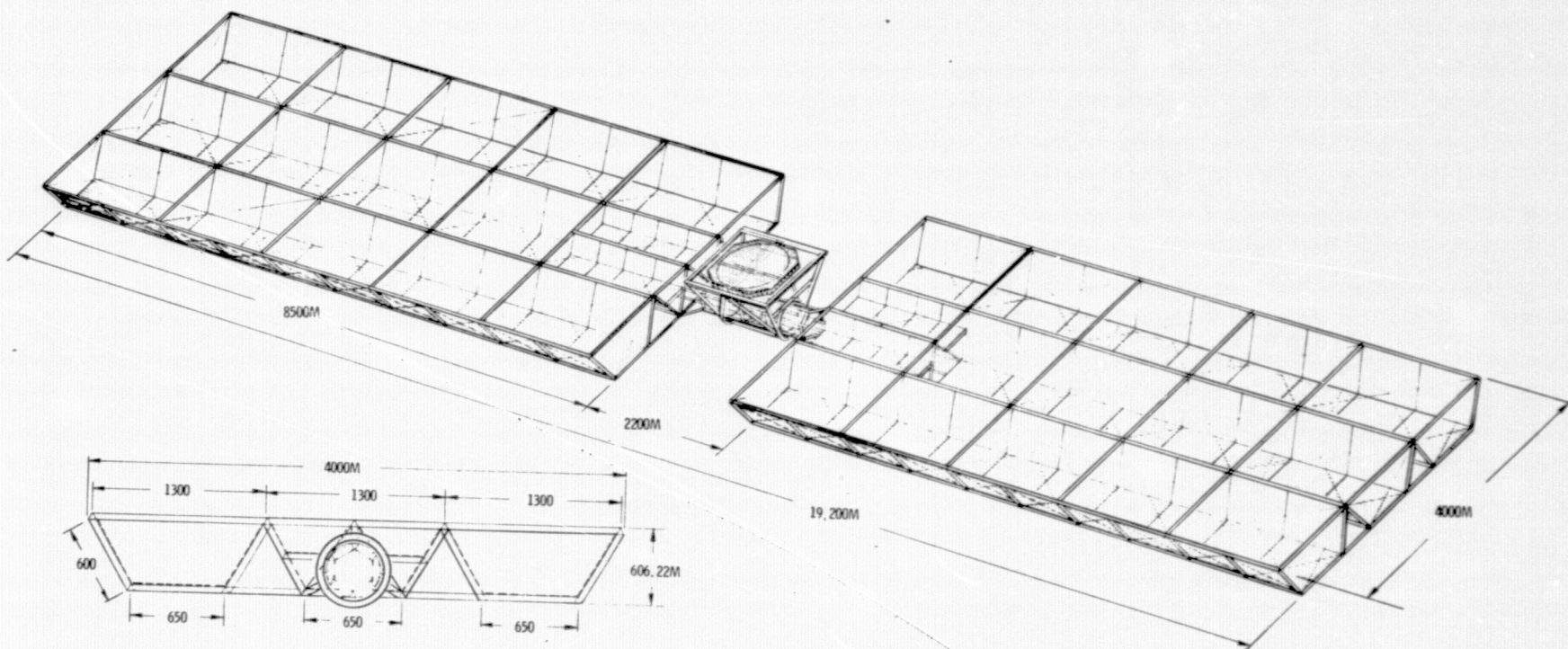


CONCLUSIONS

- SINGLE PASS CONSTRUCTION TECHNIQUE RECOMMENDED
- ANTENNA LOCATION NOT CRITICAL TO CONSTRUCTABILITY
- RECOMMEND THAT 3rd QUARTER CONSTRUCTABILITY TO BE BASED ON 3 TROUGH SATELLITE WITH CENTER-MOUNT ANTENNA



COPLANAR BASELINE CONFIGURATION



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EOTV CONSTRUCTION
LEO VS GEO

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EOTV REQUIREMENTS OVERVIEW

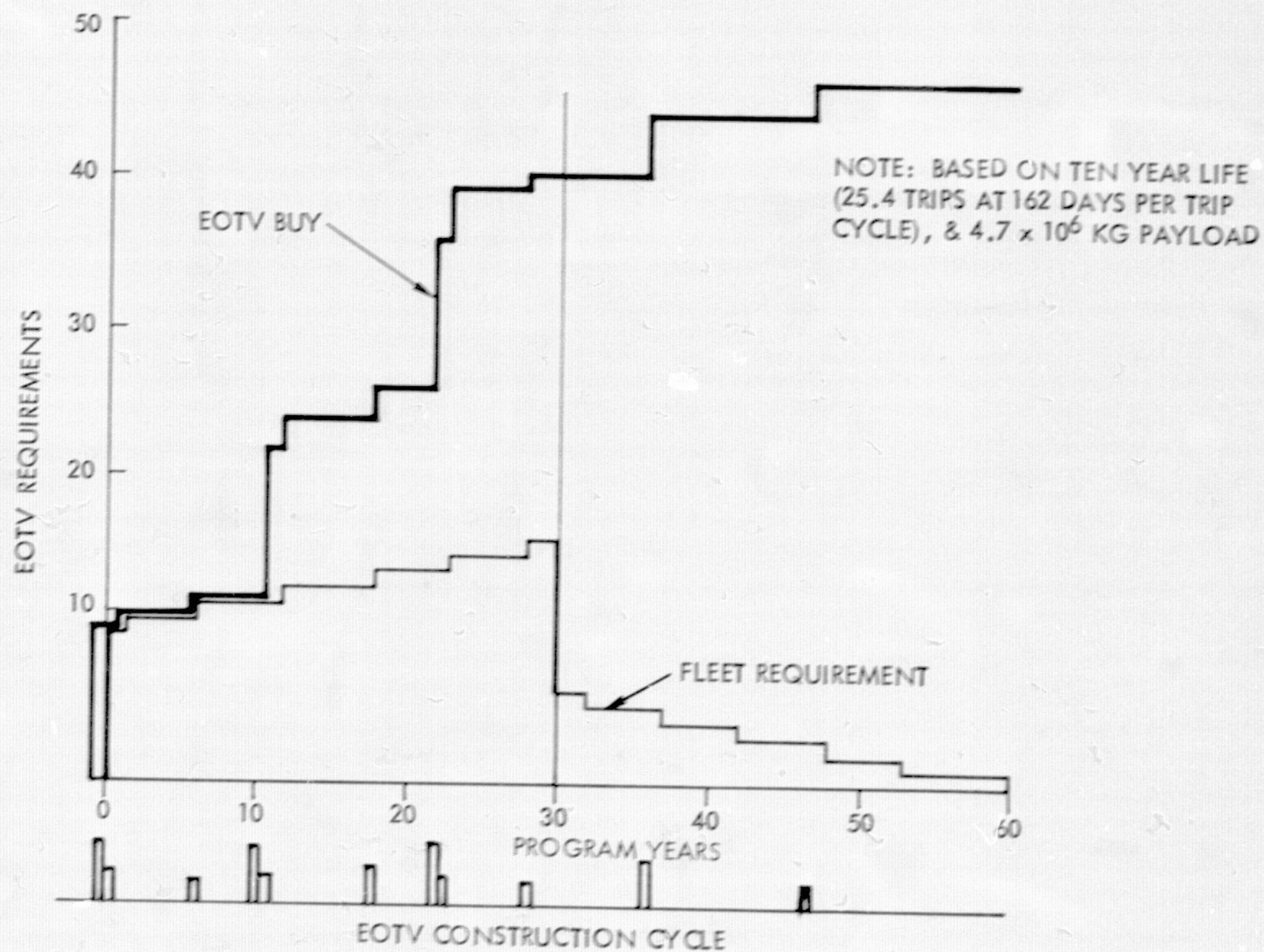
Currently, the life of an EOTV is estimated to be 10 years of operation, which translates into approximately 25 round trips between LEO and GEO. Ten EOTV's are required to transport construction material for one satellite to GEO. Because of the time required for EOTV transit, loading, and maintenance, each EOTV is limited to one trip per satellite - i.e., no "doubling up". Because of the life limitation, EOTV's must be replaced at approximately 11 year intervals.

In addition to satellite construction, maintenance material must be delivered to each satellite on an annual basis, (approximately 1×10^6 kg per satellite year).

The bottom line of the chart shows the EOTV total fleet requirements by year to transport both construction and maintenance mass to GEO. It is noted that EOTV's sufficient to support initial satellite construction are required at time zero. Since satellite construction ceases after 30 years, the requirement drops significantly to reflect only maintenance mass for the completed satellites. The downward trend is caused by decommissioning (or otherwise disposing of) satellites when they attain their design life of 30 years.

Because of EOTV life, program buy requirements exceed fleet size by a substantial margin. The top line shows these requirements. EOTV replacements fall into a total of seven build cycles (excluding initial build) as shown on the horizontal bar at the bottom of the chart. Five of these cycles occur during the first 30 years.

EOTV REQUIREMENTS OVERVIEW



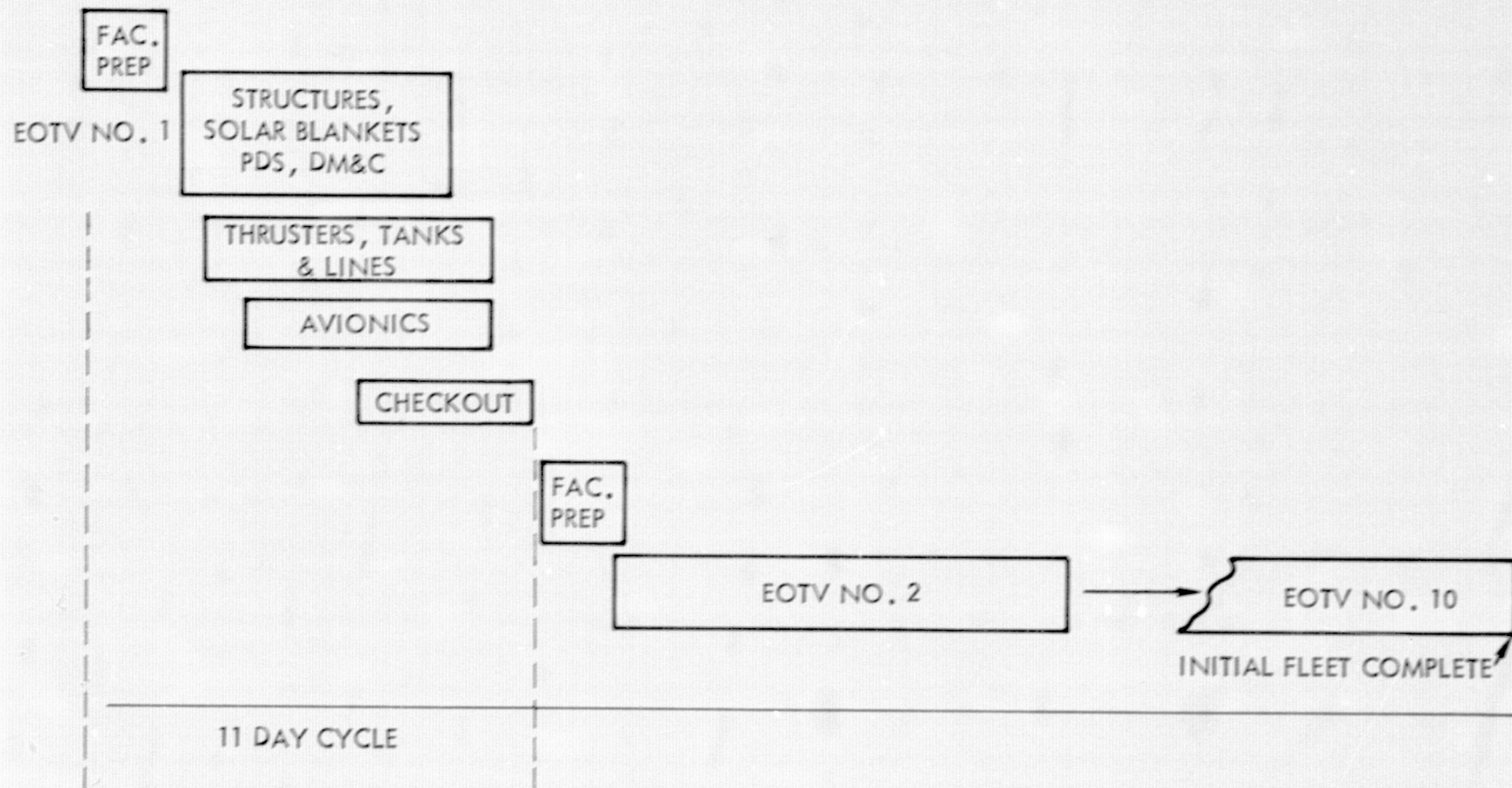
EOTV CONSTRUCTION TIMELINE

The EOTV consists of only 4 bays. Therefore, a total build cycle of seven days has been postulated. Much of the operation can be conducted in parallel as shown. It is estimated that an EOTV can be constructed in 11 days, including 2 days for facility preparation (i.e., loading beam machines, solar blanket dispensing, etc.). The time allocated for the designated activities is consistent with that allocated to satellite construction for similar activities.

EOTV CONSTRUCTION TIMELINE

DAYS

0 5 10 15 20 100 105 110



EOTV CONSTRUCTION CONCEPTS

Three potential options for constructing the EOTV's have been identified. These are:

Option 1 - Construct and maintain EOTV's in LEO with an integrated facility.

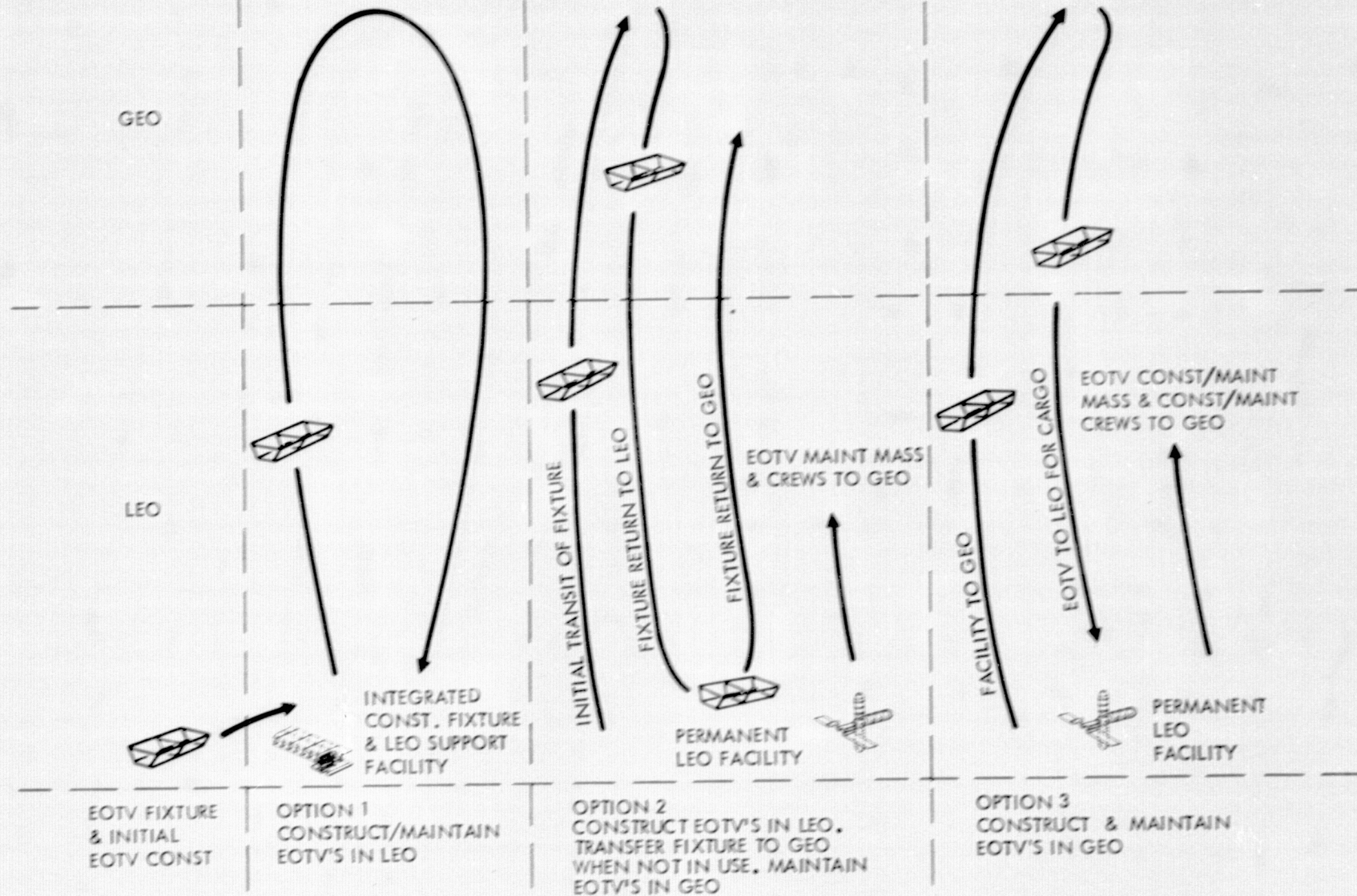
Option 2 - Construct EOTV's in LEO, but place the fixture in GEO during periods of non-building. Maintain the EOTV's in GEO except for the five EOTV build cycles, when maintenance also would be conducted in LEO. Maintain a permanent LEO facility for crew/cargo processing.

Option 3 - Transfer the fixture to GEO. Build and maintain all EOTV's in GEO. Maintain a permanent LEO facility for crew/cargo processing.

These options are depicted on the chart. It is noted that in each case, the fixture and initial EOTV are constructed in LEO.

In Option 1, no EOTV construction/maintenance crews or material are transported to GEO. Option 2 requires transportation of maintenance crews and material to GEO for 27 of the 30 program years. The other 3 years are spent in LEO, constructing EOTV's (5 build cycles). In Option 3, construction/maintenance crews and material must be transported to GEO. Additionally, each constructed EOTV must return to LEO for initial loading, a loss of about 140 days.

EOTV CONSTRUCTION OPTIONS

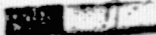


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EOTV CONSTRUCTION CONSIDERATIONS

CONSIDERATION	OPTION 1	OPTION 2	OPTION 3
• FACILITY & FIRST EOTV BUILT IN LEO	✓	✓	✓
• FACILITY RELOCATION TO GEO REQ'D		5 RT'S	1 ONE-WAY TRIP
• ALL EOTV'S MUST BE IN LEO FOR LADING	✓	✓	✓
• BENIGN ENVIRONMENT FOR EVA	✓	✓	
• MINIMUM LIGHT/DARK CYCLES DURING CONST			✓
• COMPLETED EOTV IN LOCATION FOR LADING	✓	✓	

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30 YR PROPELLANT REQUIREMENTS

Options 1, 2 and 3 are compared relative to additional propellants required over a 30 year period and resultant additional HLLV flights to carry the propellant to LEO. An additional option, 2A, has been added. This option entails constructing EOTV's in the nominal LEO orbit of 500 km and transferring the facility to a 556 km orbit between EOTV build periods to reduce stationkeeping propellant requirements. The total propellants over a 30 year period, including ΔV requirements for 5 orbital changes, is somewhat less than that required to maintain the facility permanently at 500 km.

The data displayed on the chart indicate that severe propellant (and HLLV) penalties are associated with Options 2 and 3.

Options 1 and 2A are essentially the same relevant to propellant usage, the difference being reduction of HLLV flights by one in favor of Option 2A, although operationally Option 2A is slightly more complex.

EOTV CONSTRUCTION ORBIT 30-YR PROPELLANT REQUIREMENTS

ITEM	CONSTRUCTION ORBIT	OPTION 1 LEO	OPTION 2 LEO/GEO	OPTION 2A LEO/LEO	OPTION 3 GEO
• FACILITY STATION KEEPING - LEO		622,000 KG	KG	411,040 KG	- KG
• FACILITY STATION KEEPING - GEO		-	5,940	-	6,600
• PROPELLANT FOR FACILITY TRANSFER TO GEO		-	587,500	-	117,500
• PROPELLANT FOR FACILITY TRANSFER TO LEO		-	587,500	-	-
• PROPELLANT FOR EOTV CONST MAT'L TO GEO		-	-	-	4.84×10^6
• PROPELLANT FOR EOTV MAINT MAT'L TO GEO		-	20.8×10^6	-	22.5×10^6
	PROPELLANT HLLV FLTS	622,000 KG 3	22×10^6 KG 98	411,040 KG 2	27.5×10^6 KG 123
• HLLV FLTS-EOTV MAINT CREWS - 2 HLLV'S PER ROTATION		-	108	-	120
• HLLV FLTS-EOTV CONST CREWS - 5 BUILD CYCLES AT 5 HLLV'S EACH		-	-	-	25
	TOTAL HLLV FLTS	3	206	2	266

- OPTION 1. CONSTRUCT AND MAINTAIN EOTV'S IN LEO (500 KM). STORE CONSTRUCTION FACILITY IN LEO (500 KM) WHEN NOT IN USE
- OPTION 2. CONSTRUCT EOTV'S IN LEO (500 KM) - TRANSFER CONSTRUCTION FACILITY TO GEO FOR STORAGE WHEN NOT IN USE MAINTAIN EOTV'S IN GEO.
- OPTION 2A. CONSTRUCT EOTV'S IN LEO (500 KM) - TRANSFER CONSTRUCTION FACILITY TO HIGHER LEO (556 KM) FOR STORAGE WHEN NOT IN USE
- OPTION 3. CONSTRUCT AND MAINTAIN EOTV'S IN GEO. STORE CONSTRUCTION FACILITY IN GEO WHEN NOT IN USE

CONCLUSIONS

- OPTIONS 2 AND 3 REQUIRE EXCESSIVE PROPELLANT WHEN COMPARED TO THE OTHER OPTIONS
- OPTIONS 1 AND 2A ARE COMPARABLE RELEVANT TO PROPELLANT REQUIREMENTS
- OPTION 2A REMOVES THE FACILITY FROM THE MORE HEAVILY POPULATED 500 KM ORBIT WHEN NOT IN USE
- OPTION 2A RECOMMENDED



BRIEFING OUTLINE

EXECUTIVE SUMMARY

G. M. HANLEY

REFERENCE CONCEPT CHARACTERISTICS

A. A. NUSSBERGER
R. E. OGLEVIE

REFERENCE CONCEPT CONSTRUCTION

R. E. SEXTON

MICROWAVE TRANSMISSION SYSTEM

G. D. O'CLOCK

TRANSPORTATION SYSTEM

R. P. BERGERON

EXPERIMENT/ VERIFICATION PROGRAM

W. R. RHOTE



MPTS ARRAY POINT DESIGN

The antenna parameters and radiation characteristics of the antenna array assemblies that constitute the microwave power transmission subsystem (MPTS) are investigated via Taylor Distribution. Existing data are derived from investigation of Gaussian Distribution. Preliminary results are presented here as a first step in upcoming comparison with results from Gaussian distribution considerations.

MPTS ARRAY POINT DESIGN

TAYLOR DISTRIBUTION

RADIATION CHARACTERISTICS



TAYLOR WEIGHTING

The utilization of Taylor distribution in the determination of power density as a function of antenna radius and far-field pattern consists of investigation of:

- a) Aperture distribution as defined by Taylor.
- b) Power density as a function antenna radius, determined by the Taylor distribution function.
- c) Far-field pattern as determined by Taylor distribution.

TAYLOR WEIGHTING

APERTURE DISTRIBUTION

POWER DENSITY vs RADIUS DETERMINATION

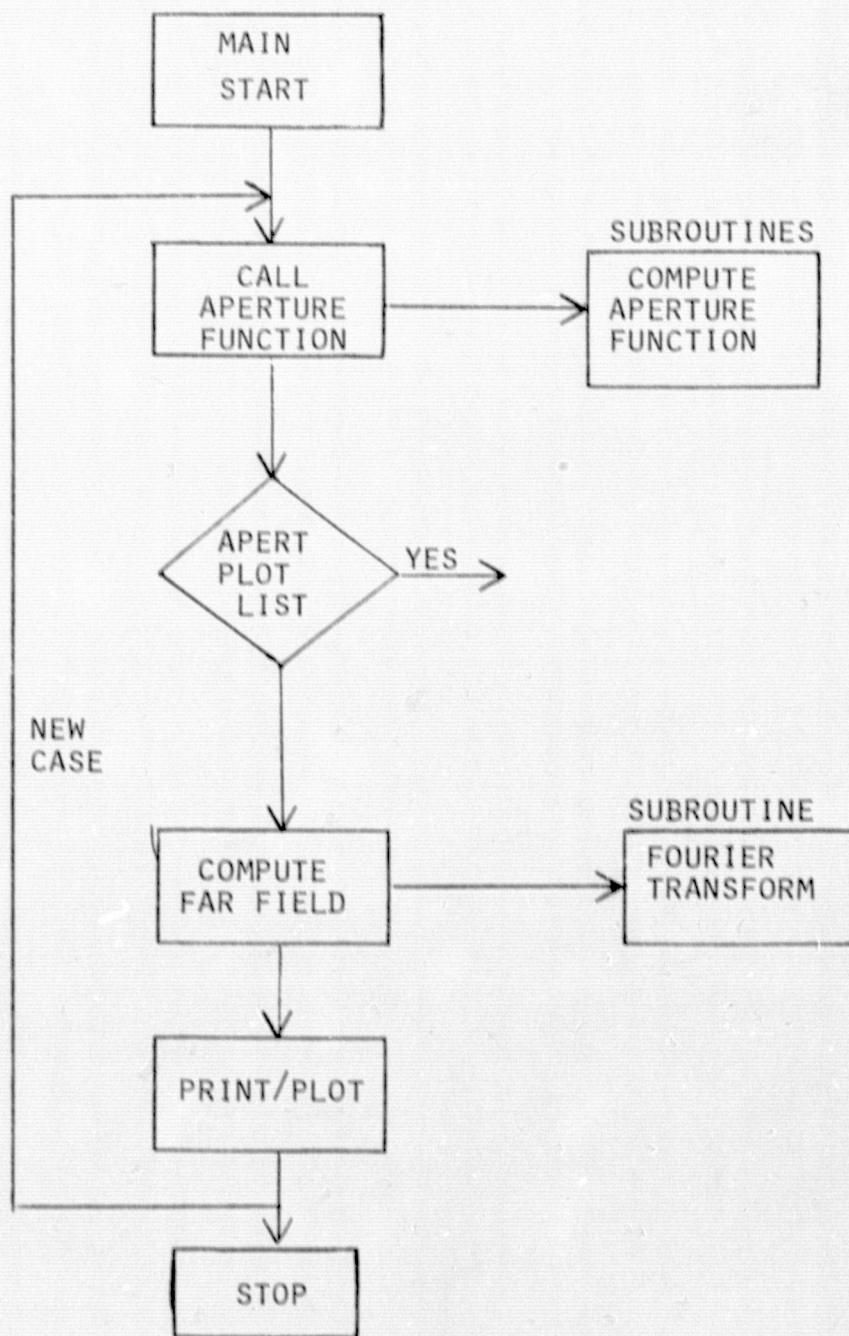
FAR FIELD PATTERN



COMPUTATIONAL PROGRAM

The objective is to develop computational tools to handle antenna calculations in a most general fashion. A computer program is being developed to have a great deal of flexibility in handling any antenna aperture function. A number of aperture functions can be stored as subroutines. The far field pattern is calculated using the appropriate aperture function.

COMPUTATIONAL PROGRAM



APERTURE DISTRIBUTION

The Taylor aperture distribution is explicitly written out to show the functional dependence of various parameters.

APERTURE DISTRIBUTION

$$G(\rho) = \sum_{m=0}^{\bar{n}-1} \frac{2 F(\mu_m, A, \bar{n})}{[J_0(\pi \mu_m)]^2 \pi^2} J_0(\mu_m \rho)$$

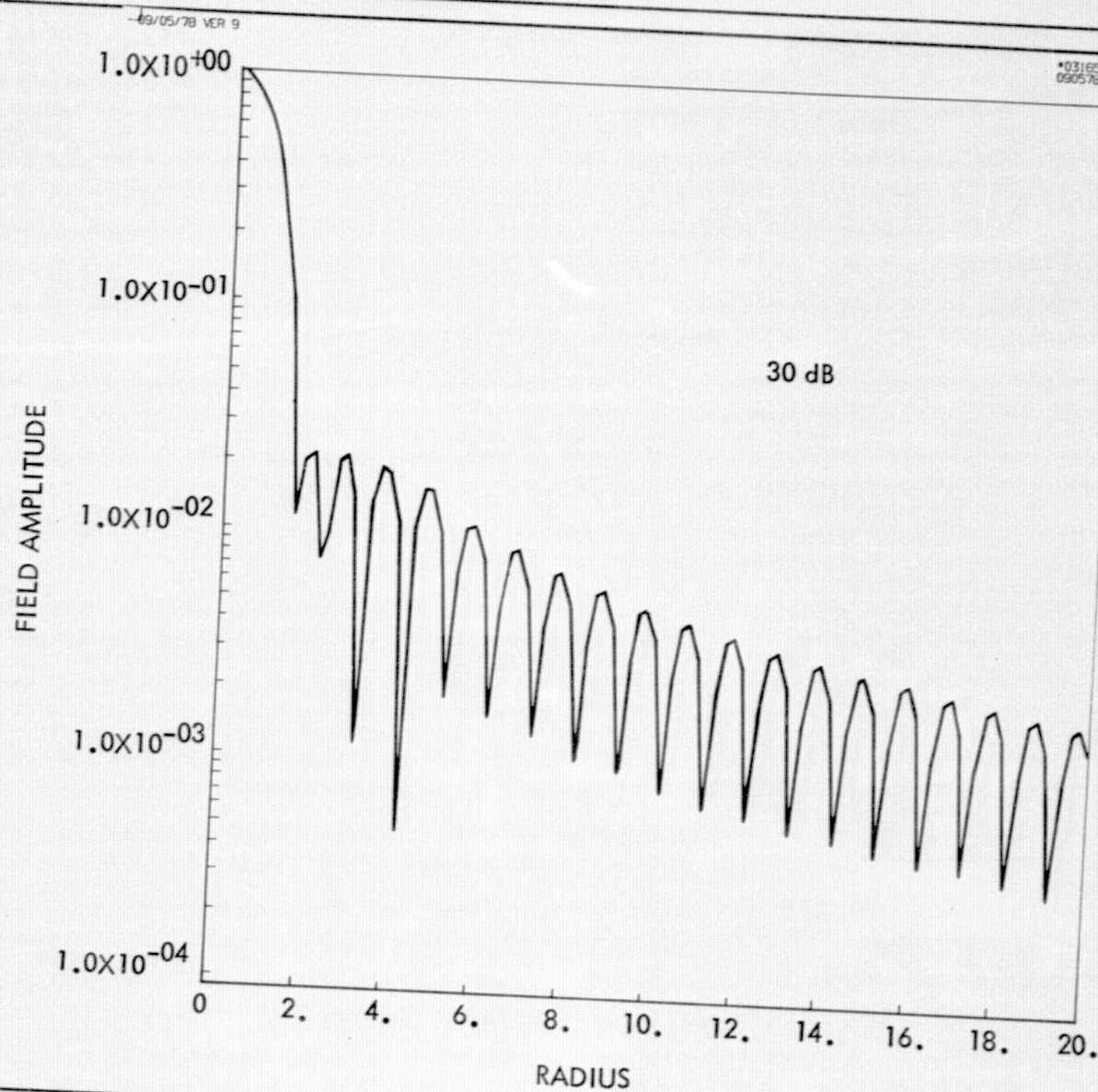
$$F(\mu_m, A, \bar{n}) = \begin{cases} 1 & m = 0 \\ -J_0(\pi \mu_m) \frac{\frac{\bar{n}-1}{11} 1 - \frac{\mu_m^2}{\tau^2 [A^2 + (n-1/2)^2]}}{\frac{\bar{n}-1}{11} 1 - \frac{\mu_m^2}{\mu_n^2}} & 0 < m < \bar{n} \\ 0 & \begin{matrix} n = 1 \\ n \neq m \end{matrix} \quad m \geq \bar{n} \end{cases}$$



FIELD PATTERN DISTRIBUTION

The far field pattern is obtained by performing a Fourier transform of the Taylor Distribution function. The case presented is a 30 db sidelobe level with $\bar{n} = 5$. The field is normalized and plotted against the normalized radius.

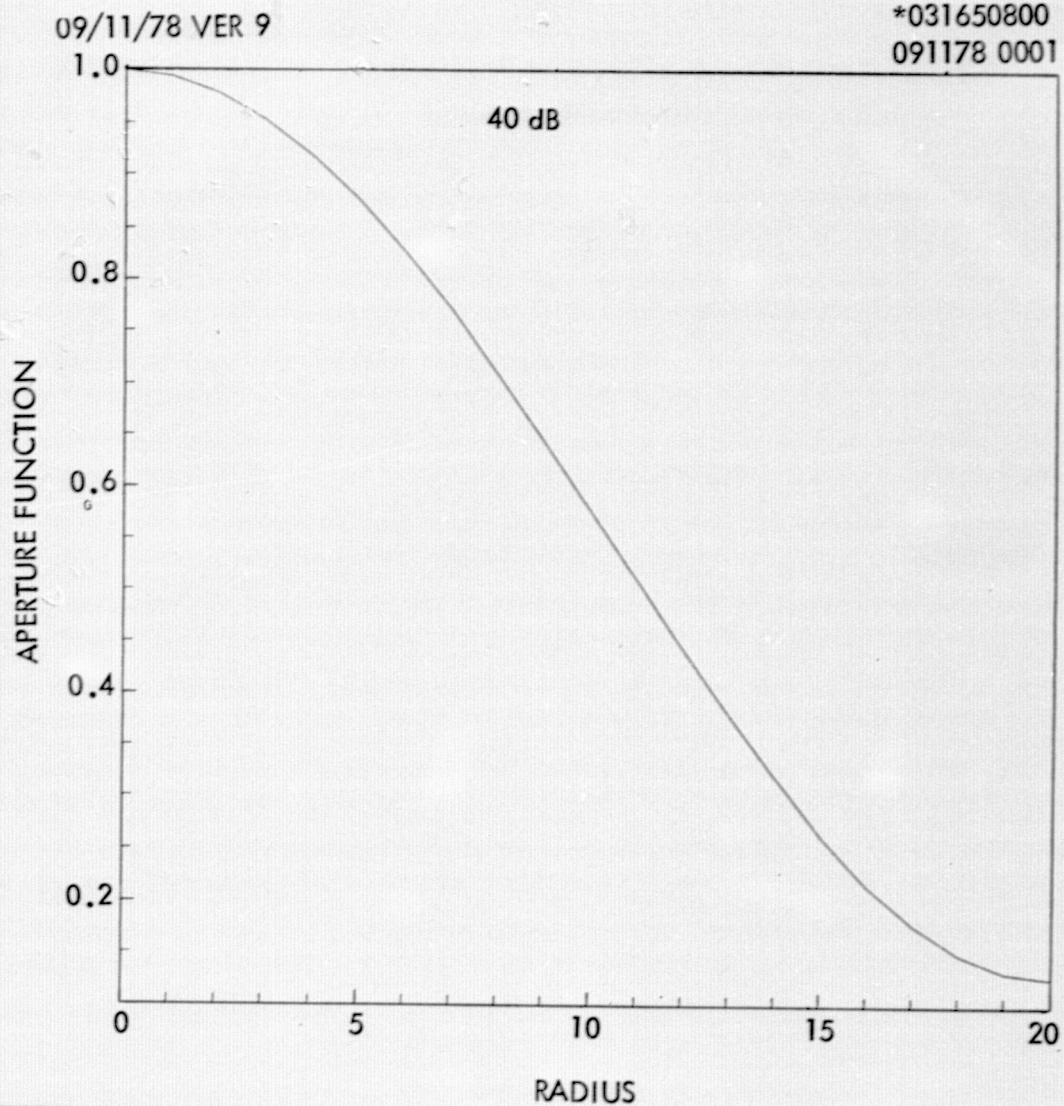
FIELD PATTERN DIST. (NBAR = 5) (NORMALIZED)



APERTURE DISTRIBUTION

The following four plots are the normalized Taylor distribution function plotted as a function of normalized radius. The four cases are 25, 30, 35, and 40 db side-lobe levels and $\bar{n} = 5$.

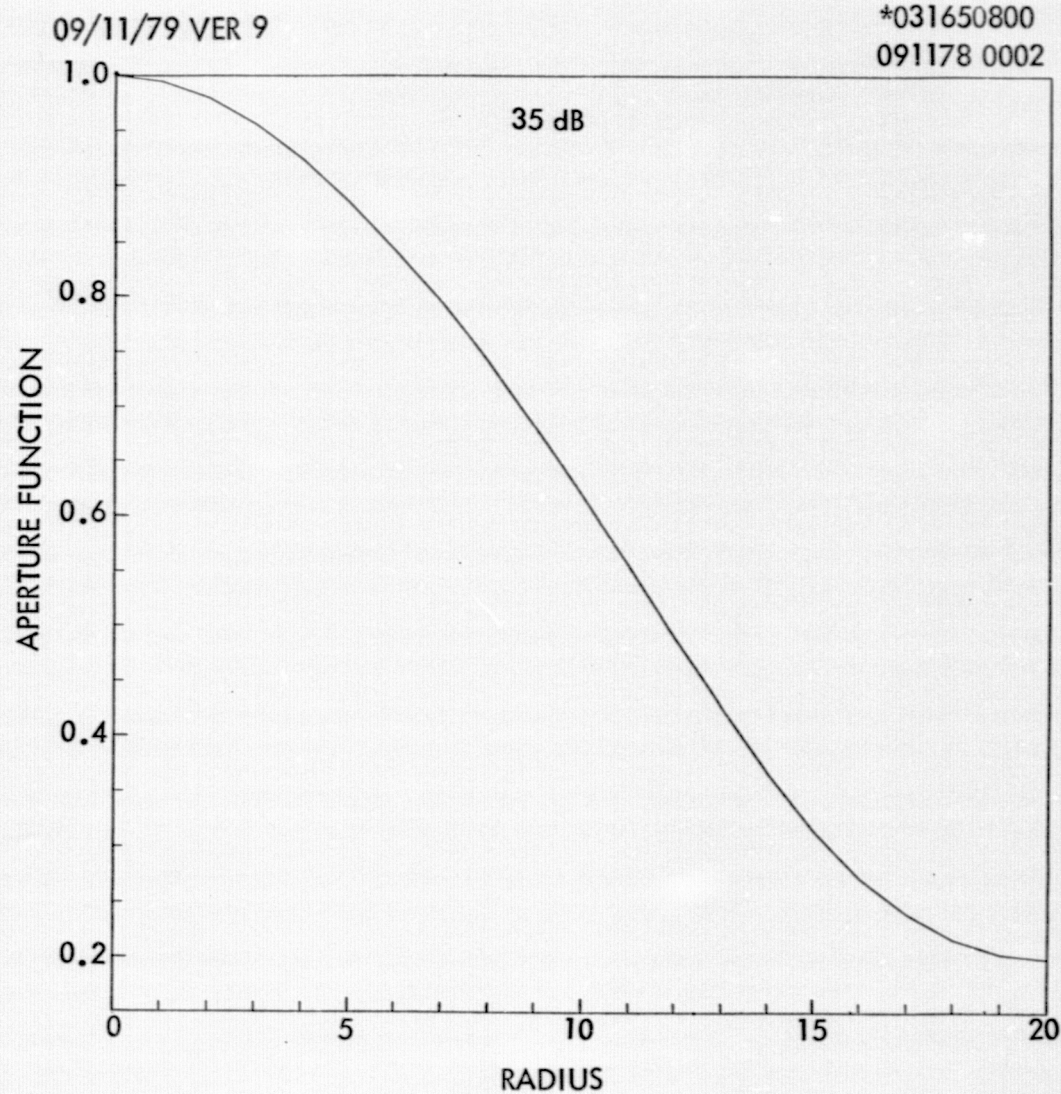
APERTURE DIST. (NBAR = 5)
(NORMALIZED)



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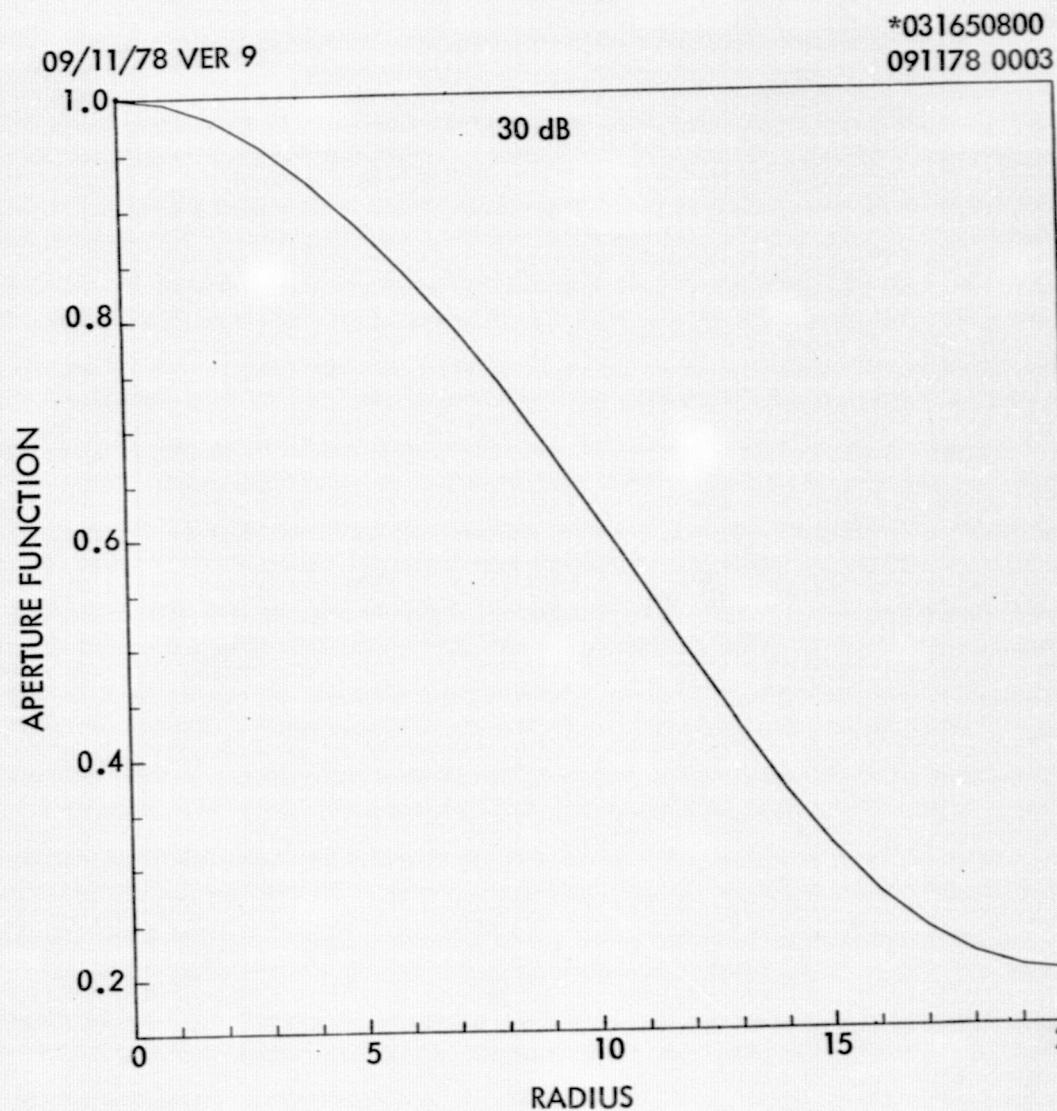
APERTURE DIST. (NBAR = 5)
(NORMALIZED)



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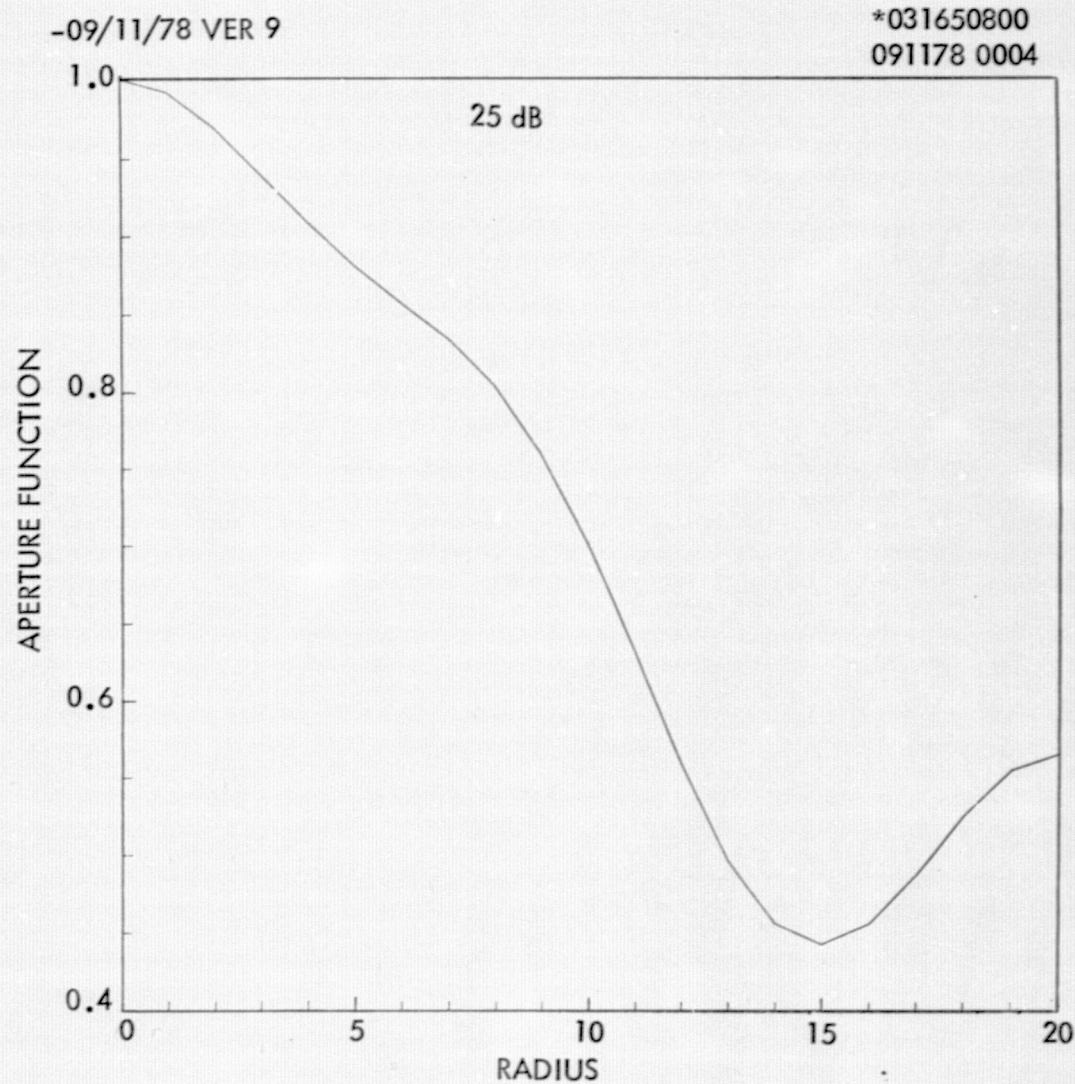
APERTURE DIST. (NBAR = 5)
(NORMALIZED)



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APERTURE DIST. (NBAR = 5)
(NORMALIZED)



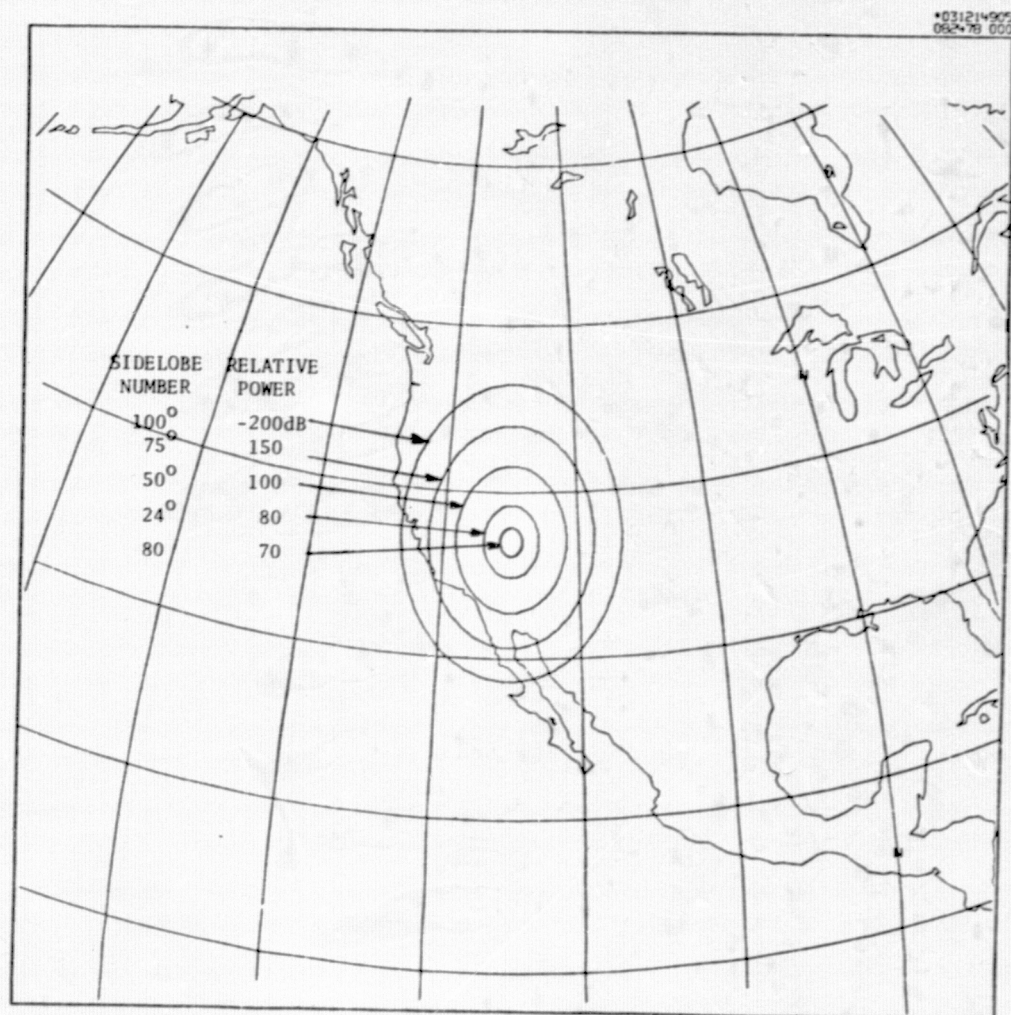
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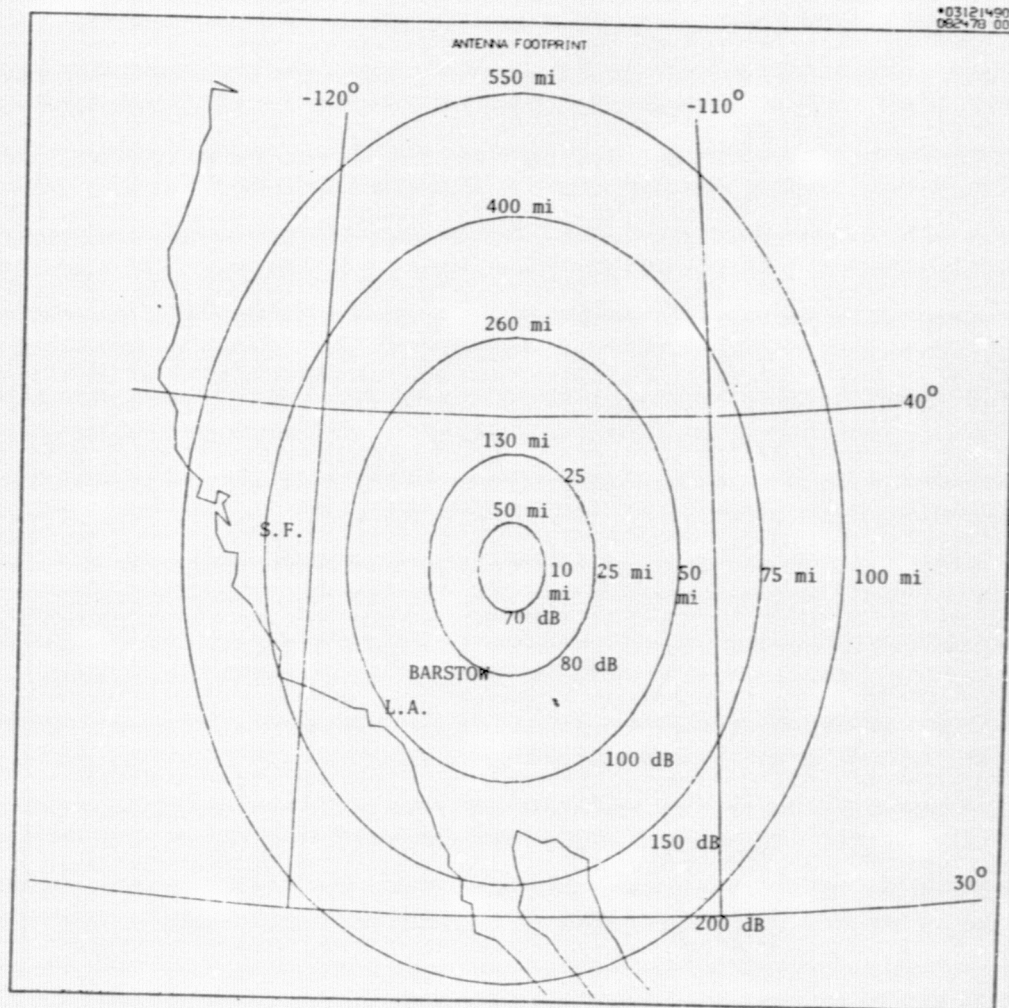
ANTENNA FOOTPRINT

The antenna footprint (next two charts) from an SPS satellite antenna is presented. The concentric ellipses represented certain sidelobe levels and power levels as labeled. The dimensions (miles) of the sidelobes are also marked to indicate the distance of the sidelobe from the center.

ANTENNA FOOTPRINT



EARTH ILLUMINATION FROM SPS EQUATORIAL SATELLITE ANTENNA



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COMPARISON OF SOLID STATE AND KLYSTRON
APPROACHES FOR SPS MPTS

This chart presents a comparative picture of the transistorized solid state and the klystron dc-RF converter in terms of their estimated lifetimes, high volume production, operation, and maintainability. The specifics are self-explanatory.

COMPARISON OF SOLID STATE AND KLYSTRON APPROACHES FOR SPS MPTS

	<u>SOLID STATE (TRANSISTOR)</u>	<u>KLYSTRON</u>
● LIFETIME	LIFETIME POTENTIAL IS MUCH HIGHER FOR THE TRANSISTOR POWER MODULE.	CATHODE DEGRADATION AND HIGH VOLTAGE POWER SUPPLY BREAKDOWN LIMIT LIFETIME.
● HIGH VOLUME PRODUCTION	THE SMALL SIZE OF THE SOLID STATE POWER MODULE PROVIDES MANUFACTURING ADVANTAGES AND REPRODUCIBILITY POTENTIAL.	REQUIRES HIGH TOLERANCES. LARGE SIZES INVOLVED. SIGNIFICANT HIGH VOLUME TEST PROBLEMS.
● OPERATION	CAN BE SLOWLY SWITCHED ON AND OFF WITHOUT DEGRADATION.	PARTS OF KLYSTRON MUST REMAIN "ON" AT ALL TIMES.
● MAINTAINABILITY	NO X-RAY PROBLEMS. SMALL MODULE SIZES ALLOW EASY REPLACEMENT.	SIGNIFICANT X-RAY AND LEAKAGE PROBLEMS WHICH ARE HAZARDOUS TO PERSONNEL. SUSCEPTIBLE TO MAGNETIC DISTURBANCES. MANY INTERCONNECTS. AUTOMATED REPL. TECH. REQ.



IMPACT OF HIGHER SOLAR CELL EFFICIENCIES

This chart illustrates that if higher efficiencies are obtained in the primary power generating solar cells it may be possible to utilize other power converter candidates such as TRAPATT diode amplifiers or Electron Bombarded semiconductor amplifiers. However the need for low primary input voltages remains a primary concern.

IMPACT OF HIGHER SOLAR CELL EFFICIENCIES

- ALLOWS A WIDER RANGE OF SOLID STATE MICROWAVE POWER CONVERTER CANDIDATES THAT DO NOT POSSESS THE HIGH EFFICIENCY POTENTIAL THAT THE TRANSISTOR HAS.
- SOME OF THESE CANDIDATES OPERATE AT HIGHER VOLTAGE LEVELS COMPARED WITH TRANSISTORS:

<u>DEVICE</u>	<u>EFFICIENCY</u>	<u>SUPPLY VOLTAGE</u>	<u>OUTPUT POWER</u>
TRAPATT DIODE AMPLIFIER	≤50%	≤150V	20W
ELECTRON BOMBARDED SEMI- CONDUCTOR (EBS) AMPLIFIER*	<45%	15KV @ 10 mA 60V @ 100A	>3KW

- THE VOLTAGES ASSOCIATED WITH THE POWER HANDLING CAPABILITIES OF EACH DEVICE ARE STILL TOO LOW TO REDUCE THE DC/DC CONVERTER DESIGN PROBLEM.

*VERY SMALL SIZE AND LIGHT WEIGHT.



TRANSISTOR COLLECTOR BREAKDOWN VOLTAGE (V_{BC})

This chart illustrates the functional relationship between the base width (W_B), epitaxial width (W_{EPI}) and doping concentration (N_A , N_O) of the various types of transistor configurations investigated.

TRANSISTOR COLLECTOR BREAKDOWN VOLTAGE (BV_C)

- THEORETICALLY, THE BASIC FORM OF BV_C SHOWS A FUNCTIONAL RELATIONSHIP BETWEEN BASE WIDTH (W_B), EPITAXIAL WIDTH (W_{EPI}) AND DOPING CONCENTRATION (N_A , N_D). IN GENERAL:

$$BV_C = \frac{e N_A W_B^2}{2 \epsilon_r \epsilon_0} \quad (\text{UNIFORM BASE ALLOY TRANSISTOR})$$

$$BV_C = \frac{e}{\epsilon_r \epsilon_0} \left[N_A W_{EPI} W_B - \frac{N_D W_{EPI}^2}{2} + \frac{N_A W_B^2}{2} \right] \quad (\text{DOUBLE DIFFUSED TRANSISTOR})$$

- FOR OPTIMUM CONDITIONS WITH THE DOUBLE DIFFUSED TRANSISTOR:

$$W_{EPI} (\text{OPT}) = \frac{N_A W_B}{N_D}$$

$$BV_C \approx \frac{e N_A W_B^2}{2 \epsilon_r \epsilon_0} \quad (N_A \approx 2 \cdot 10^{19}/\text{cm}^3, N_D \approx 5 \cdot 10^{20}/\text{cm}^3)$$

- IF W_B IS APPROXIMATELY 4μ AND ϵ_r IS 12:

$$BV_C \approx 25V$$

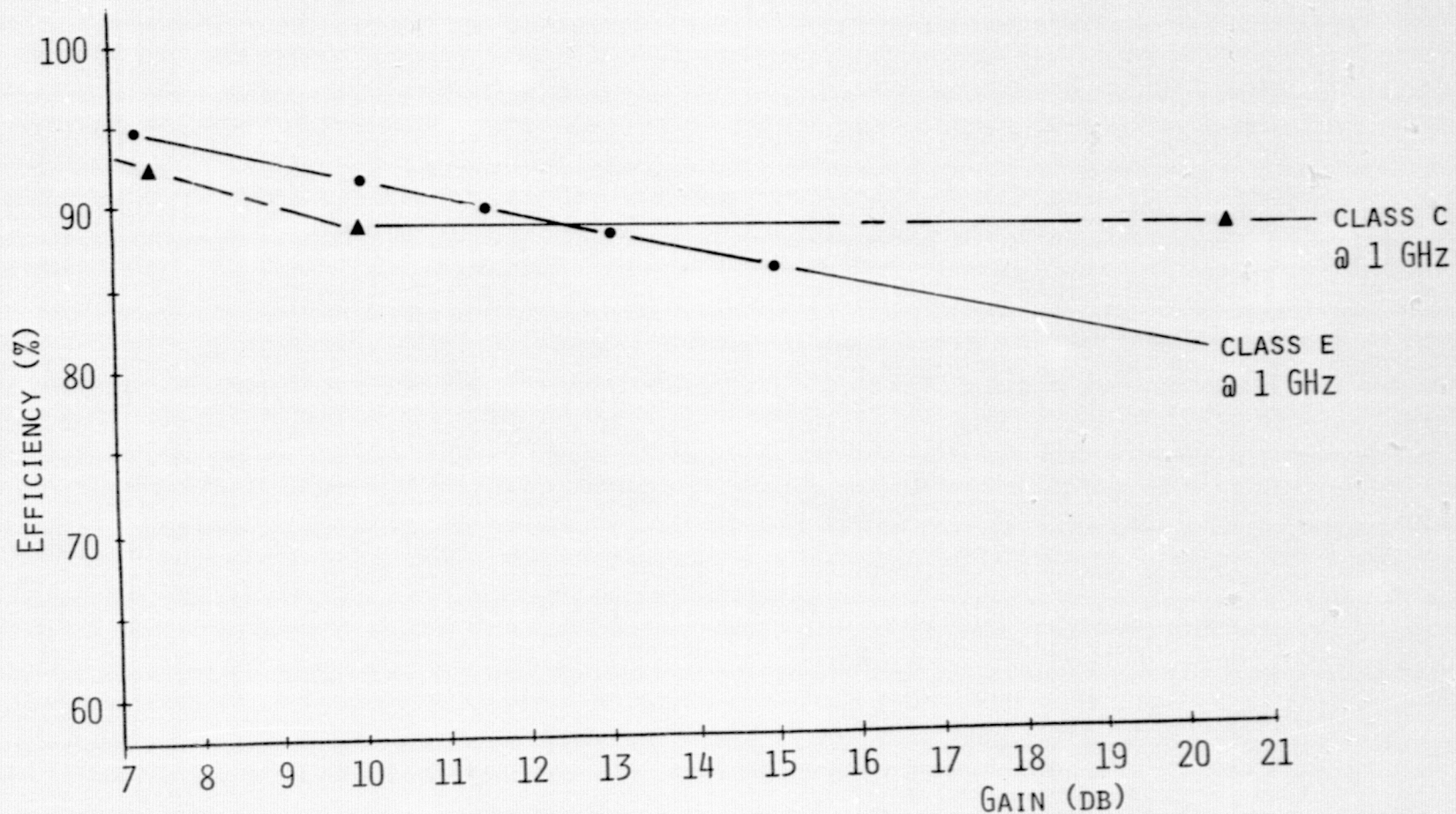
- IT IS READILY APPARENT THAT THE TRANSISTOR BREAKDOWN VOLTAGE IS A FUNCTION OF DOPING AND DEVICE STRUCTURE.



COMPARISON OF CLASS E AND CLASS C AMPLIFIER EFFICIENCIES
VS GAIN AT 1 GHz

This chart compares bipolar transistor based Class E and C amplifiers as a function of output gain. For the SPS program a gain of 15-20 dB is desired thus implying that Class C operation should be implemented.

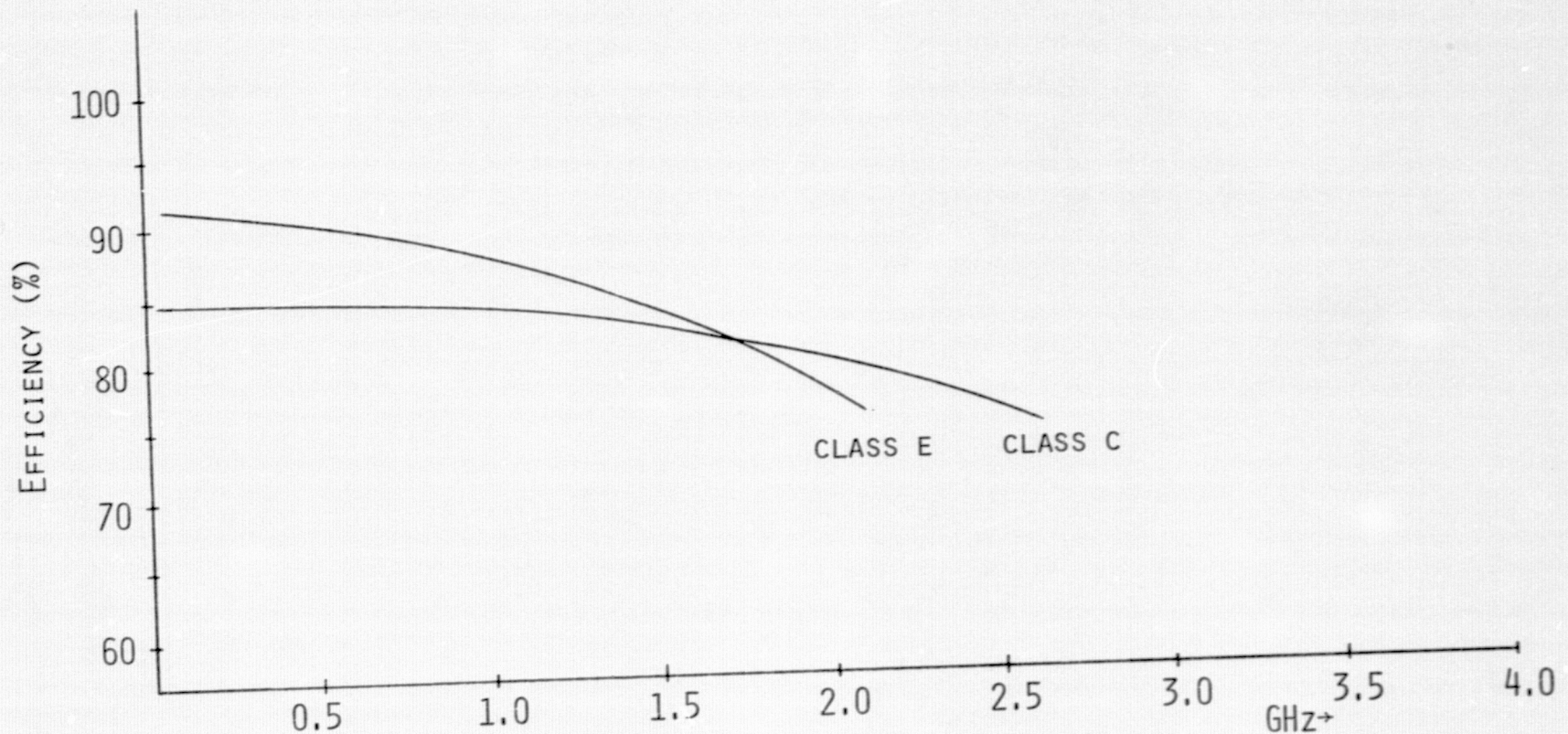
COMPARISON OF CLASS E AND CLASS C AMPLIFIER EFFICIENCIES VS. GAIN AT 1 GHZ
(FROM COMPUTER DESIGN SIMULATION OF BIPOLAR TRANSISTOR & BASIC AMPLIFIER CIRCUIT)



COMPARISON OF CLASS E AND CLASS C BIPOLAR TRANSISTOR
AMPLIFIER EFFICIENCIES VS FREQUENCY

This chart shows that if a specific gain is selected, (in this case 10 dB), the efficiency of a Class C amplifier is still more desirable at the SPS operating frequency of 2.45 GHz.

COMPARISON OF CLASS E AND CLASS C BIPOLAR TRANSISTOR AMPLIFIER EFFICIENCIES VS. FREQUENCY
(PRELIMINARY INDICATIONS FOR AN AMPLIFIER WITH 10 DB GAIN, ACCOUNTING FOR SOME CIRCUIT INEFFICIENCIES)



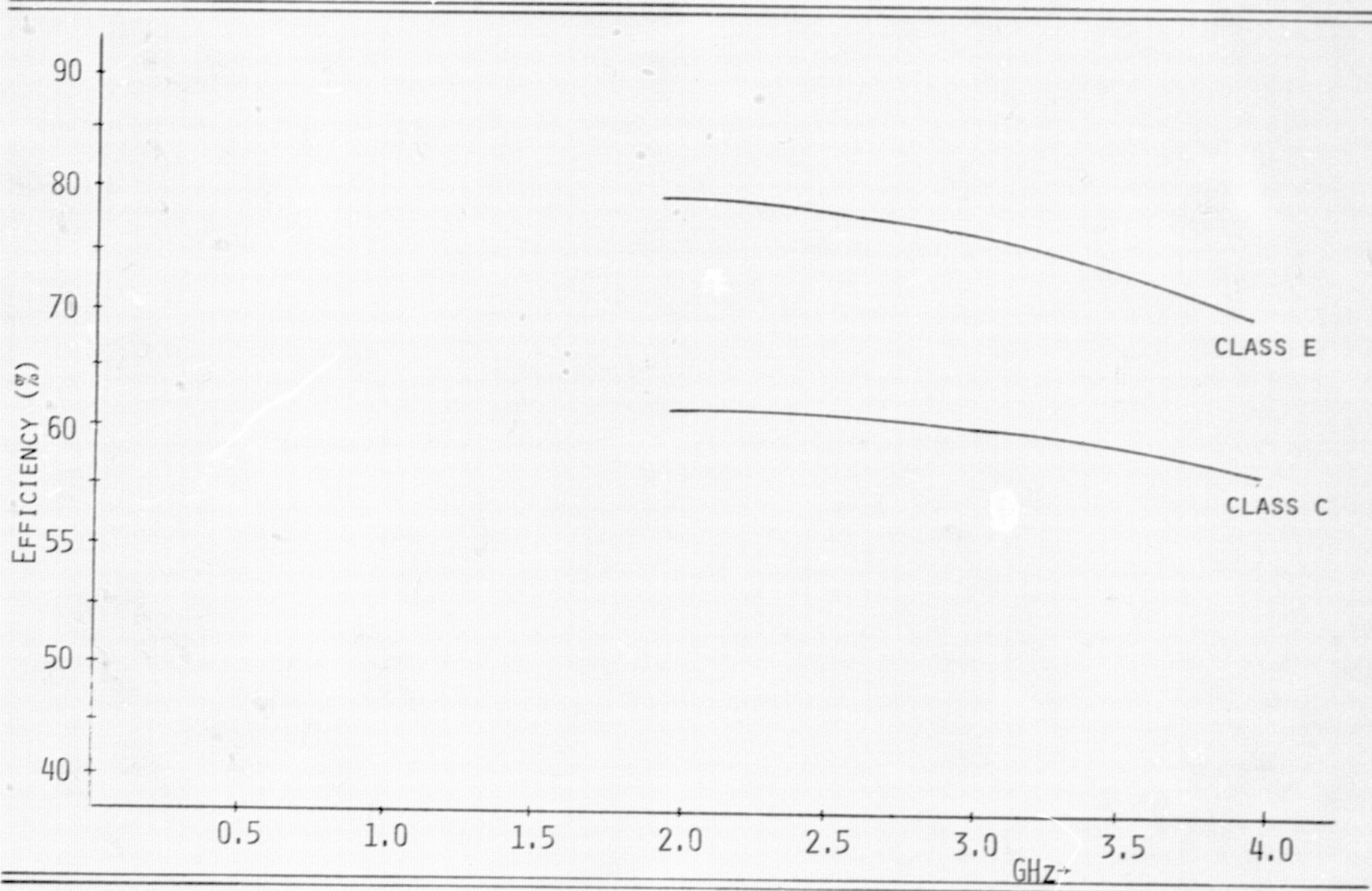
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COMPARISON OF CLASS E AND CLASS C GaAs AMPLIFIER
EFFICIENCIES VS. FREQUENCY

This chart illustrates that for a FET based transistor amplifier with a gain of 10 dB the most obvious selection, over a wide range of transmitter frequencies, would be Class E.

COMPARISON OF CLASS E AND CLASS C GaAs FET AMPLIFIER EFFICIENCIES VS. FREQUENCY
(PRELIMINARY INDICATIONS FOR AN AMPLIFIER WITH 10 DB GAIN, ACCOUNTING FOR SOME CIRCUIT INEFFICIENCIES)



ALTERNATIVE CHOICES FOR VARIOUS AMPLIFIER CANDIDATES
AT 2.45 GHz WITH GAINS OF 15 dB OR MORE BASED ON
PRESENT KNOWLEDGE

This chart is self-explanatory.

ALTERNATIVE CHOICES FOR VARIOUS AMPLIFIER CANDIDATES AT 2.45 GHZ WITH
GAINS OF 15 DB OR MORE BASED ON PRESENT KNOWLEDGE

	<u>GAAs BIPOLAR TRANSISTOR</u>	<u>GAAs FET</u>
<u>CLASS C</u>	HIGHEST EFFICIENCY (>75%) FOR GAINS ABOVE 10 DB WITH PRESENT STATE-OF-THE-ART DEVICES.	EFFICIENCY COULD BE LESS THAN 60% AT 2.5 GHZ.
<u>CLASS E</u>	EFFICIENCY DROPS OFF BEYOND 1.5 GHZ.	HIGH EFFICIENCY (>75%) FOR GAINS ABOVE 10 DB WITH PRESENT STATE-OF-THE-ART DEVICES.

DC/RF SOLID-STATE/WAVEGUIDE SPS CONCEPTS

The data presented in the following charts develop a series of ideas as to the feasibility of adapting the dc/RF, solid-state, waveguide principles, developed by Aerospace Corporation GSSPS to the MSFC four-trough gallium-arsenide SPS under study contract to Rockwell International.

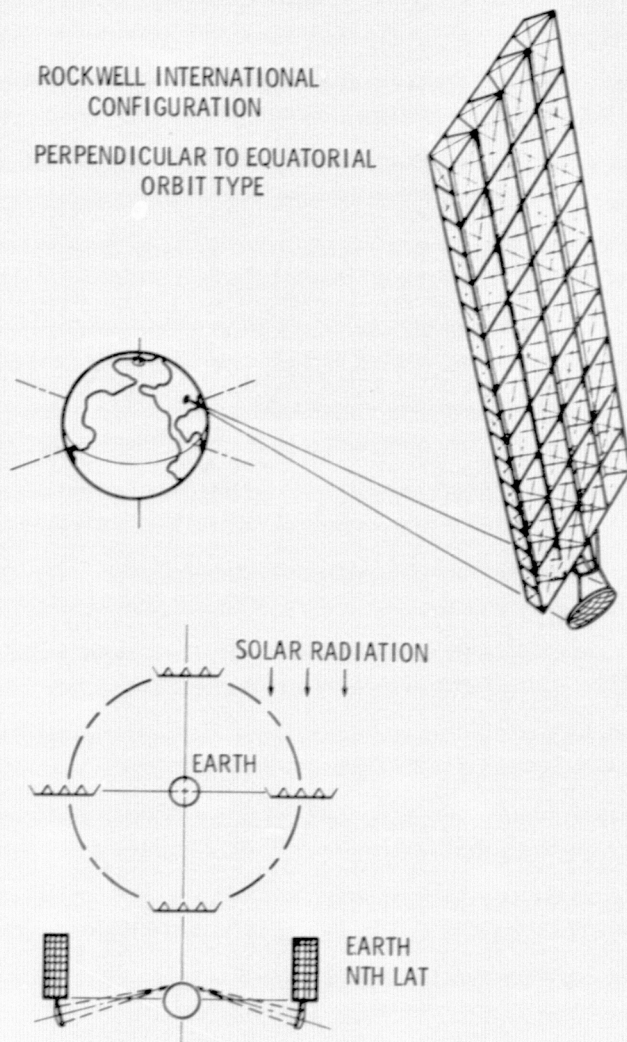
The facing chart shows the general arrangement of the Aerospace GSSPS and the Rockwell SPS systems. In the spring and fall of each year, the solar collector panels of GSSPS obscure each other. Aerospace Corporation suggests a twice-per-orbit libration about the satellite roll axis to minimize this obscuration. In the winter and summer, the panels do not obscure each other due to the 23.5-degree angle between the earth equatorial and ecliptic planes. The libration cycles must be phased to cause a once-a-year precession if synchronization with the changing sun angle is to be maintained. All of these maneuvers may require a complex attitude control system for the 47 km long string-like satellite to maintain sun-pointing of the solar panels and earth-pointing (to latitudes not on the equator) of the lens antenna.

The Rockwell configuration simplifies sun-pointing of solar cells and earth-pointing of the reflector-antenna using a simple mechanical rotary joint and pivot system containing no electrical or RF components. The satellite attitude is perpendicular to the equatorial orbit plane with the antenna located on the South Pole end of the satellite pointing toward northern earth latitudes. For south latitude antenna-pointing, the antenna is mounted on the North Pole end of the satellite. Ion propulsion units maintain active control over satellite attitude and either control-moment gyros or simple electric motor drives maintain earth-pointing of the antenna. The Aerospace Corporation GSSPS uses a three-layer (butterfly dipole/phase-shifter/butterfly dipole) lense antenna to transmit energy, emitted from a horn mounted on the end of the 47 km long main waveguide, in a coherent manner to earth. Latitude pointing and minor longitude pointing is accomplished by wave-front phasing and may require some mechanical pointing to maintain low sidelobe requirements.

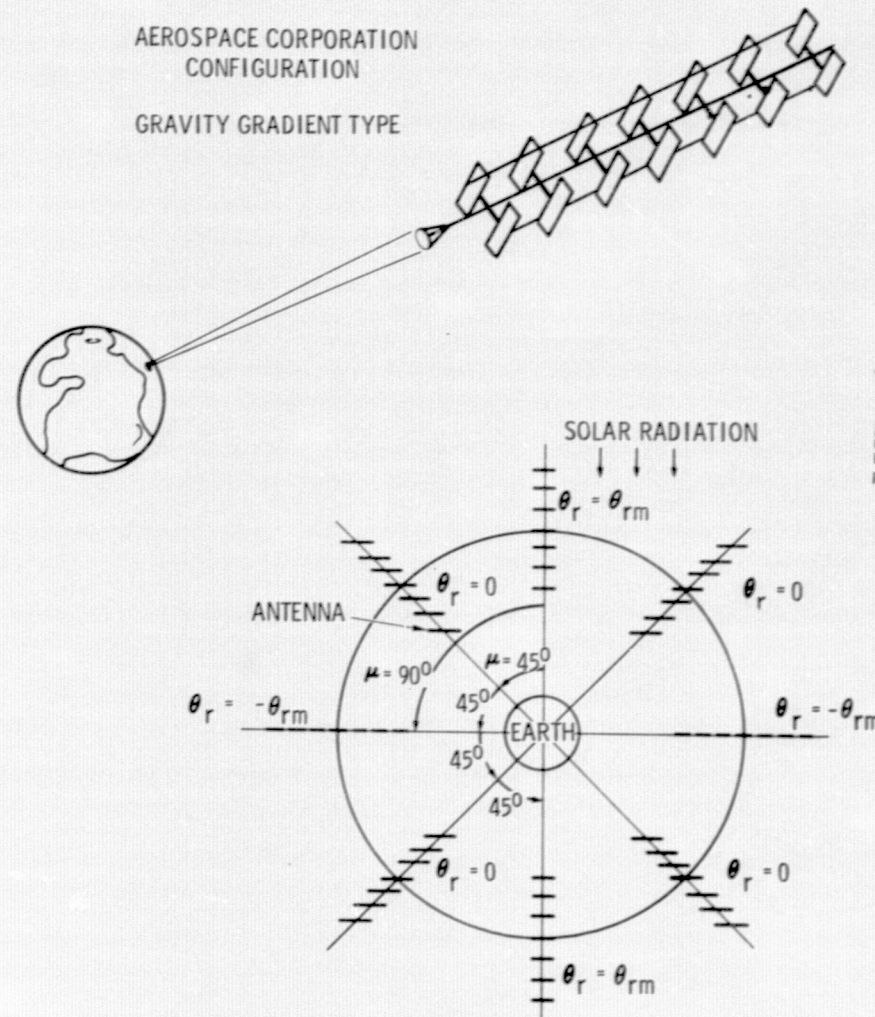
The Rockwell configuration uses a paraboloidal wire mesh antenna to reflect and collimate energy transmitted by a waveguide mounted transmitter horn that is designed as a field lens to emit a spherical wave front. The waveguide horn is located at the focal point of the paraboloidal, wire mesh mirror.

DC/RF SOLID-STATE/WAVE-GUIDE SPS CONCEPTS

ROCKWELL INTERNATIONAL
CONFIGURATION
PERPENDICULAR TO EQUATORIAL
ORBIT TYPE



AEROSPACE CORPORATION
CONFIGURATION
GRAVITY GRADIENT TYPE



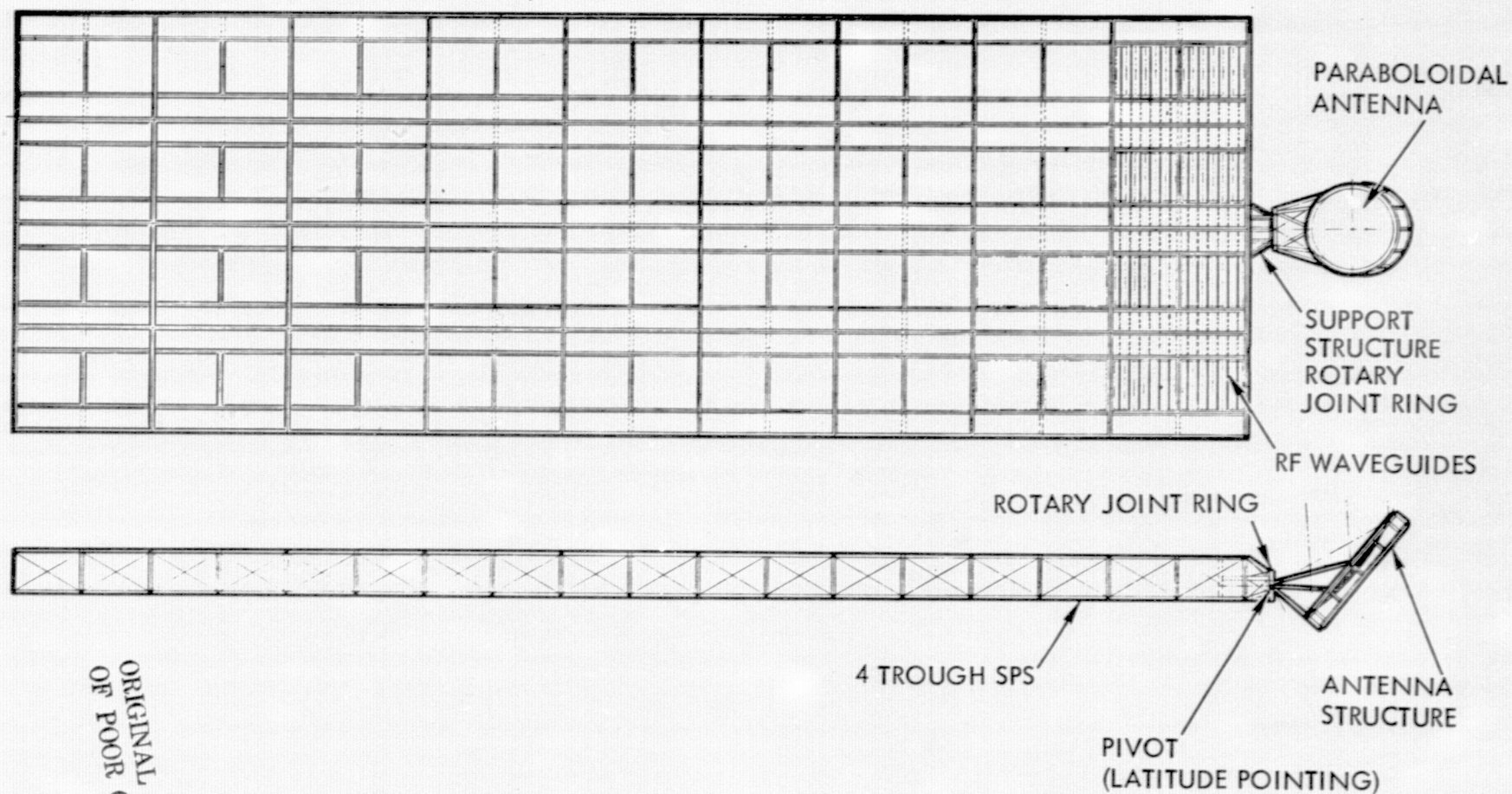
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SOLID-STATE SATELLITE POWER SYSTEM

The general configuration of the Rockwell/MSFC four-trough SPS is shown on the facing chart. An antenna support structure is attached to the 600-m centrally located triangular structure. The support structure includes a rotary ring assembly and an antenna assembly attached by pivots to the rotating element of the ring assembly. The rotating ring (axis along SPS centerline) permits the daily rotation of the antenna; the pivot permits latitude antenna pointing.

RF waveguides and solar cell panel in transverse orientation are shown.

SOLID STATE SATELLITE POWER SYSTEM



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Rockwell International
Space Division

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ROCKWELL DC/RF SOLID-STATE/WAVEGUIDE CONCEPT DETAILS

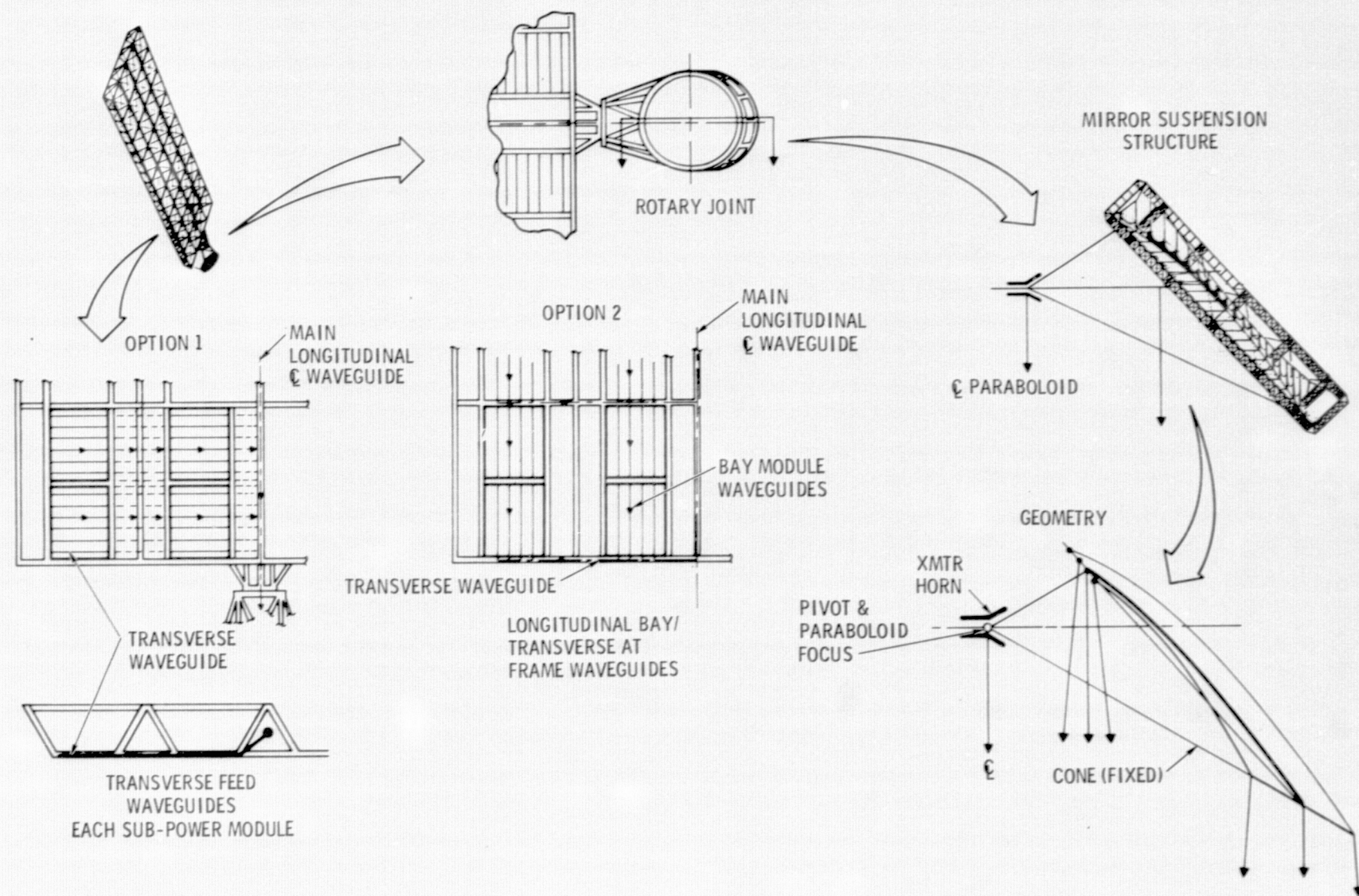
The facing chart shows the basic elements to adapt the Aerospace Corporation solid-state, waveguide RF power transfer concept.

Two options for solar cell, waveguide orientation are shown. Option 1 illustrates a transverse orientation of solar cells and RF waveguide feeds. Solar cells on each side of the waveguide generate and conduct dc electrical power to solid-state devices, mounted directly on the waveguide, that convert dc to RF power. The waveguides transmit the RF energy generated to a centrally located main waveguide where matching waveguide elements insert the lateral waveguide energy into the central waveguide connected to the transmitter horn.

Option 2 shows a longitudinal orientation of solar cells and waveguides. At the end of each structural bay, the longitudinal waveguides are connected to lateral waveguides which, in turn, are connected to the main longitudinal waveguide which feeds the transmitter horn. Power levels, waveguide matching, and system assembly procedures will determine which option is selected.

The antenna rotary joint shows the location of a section cut of the antenna illustrated at the right side of the chart. The geometry of the antenna, the relationship of the antenna horn to the antenna paraboloidal surface, and earth pointing of the collimated beam are shown below the section cut.

ROCKWELL DC/RF SOLID-STATE/WAVEGUIDE CONCEPT DETAILS



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GEOMETRY

The Aerospace Corporation lens antenna requires a diagonal mirror to permit attachment to the Rockwell SPS.

The case of a diagonal mirror pivoted at the intersection of the horn axis and the lens equatorial pointing axis is shown in the upper left-hand corner of the facing chart. For each degree the flat mirror rotates about the pivot point, the lens antenna must rotate two degrees about the pivot point remaining perpendicular to the fixed radial centerline distance. Structural rigidity and alignment tolerance appear to be the key problems.

The case of two simple pivot motions is shown in the lower left-hand corner of the facing chart. As before, each degree of diagonal mirror rotation requires two degrees of lens antenna rotation. However, the lens in this case rotates around a fixed pivot point. The two pivot points thus may be rigidly attached to each other.

The pivot motion described accounts only for latitude pointing. Daily earth pointing (longitude fixed) of the lens/mirror assembly requires a rotary joint with its axis coincident to the waveguide/horn axis, rotating once every 24 hours. The overall structural complexity, mirror flatness, and lens tolerances make implementation of either of the lens/mirror suggestions difficult.

A third, reasonably simple, option is possible. If the transmitter horn is designed as a field lens to emit energy with a spherical wavefront from the focal point of a paraboloidal mirror, the energy will be reflected parallel to the axis of the paraboloid and collimated into a flat wavefront beam. In the near field, the energy profile will tend to be uniform.

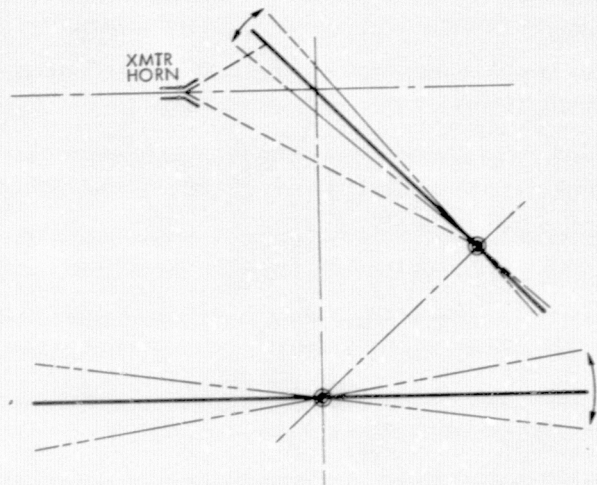
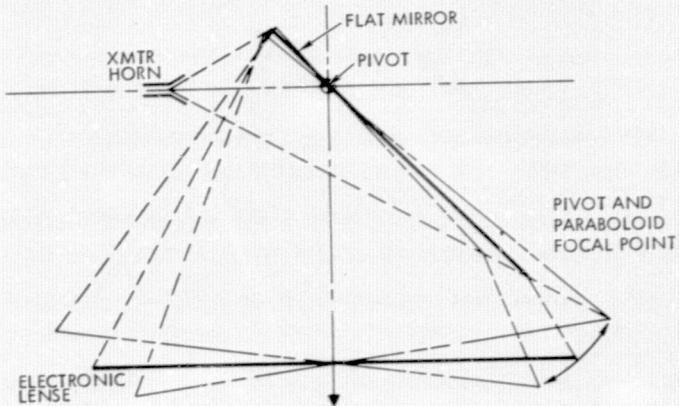
In the far field, the energy profile will approximate a Gaussian $\sin x/x$ curve. A second mirror shape potential is an ellipsoid with one focus at the transmitter horn and the second focus at the center of the rectenna located on the earth. The ellipsoidal and paraboloidal contours are very nearly identical. Other antenna contours are possible if the lens horn has an apparent index of refraction as in coherent-light optics. The mirror index of minus one may be equated to the positive lens index to effect cancellation of mirror frequency aberrations of the system (if physical optics equivalence holds true). Wave shaping by the lens may also be used to correct for mirror surface anomalies to some degree. At issue is—How much?

To effect beam pointing for daily motion, a rotary joint with axis coincident to the waveguide and transmitter horn axis is required. The plane of rotation is located at the paraboloid focal point. To effect beam latitude pointing a pivot axis in the plane of the rotary joint is required. These two axes allow the paraboloid centerline to be pointed at any required angle to place the transmitted beam at the earth rectenna.

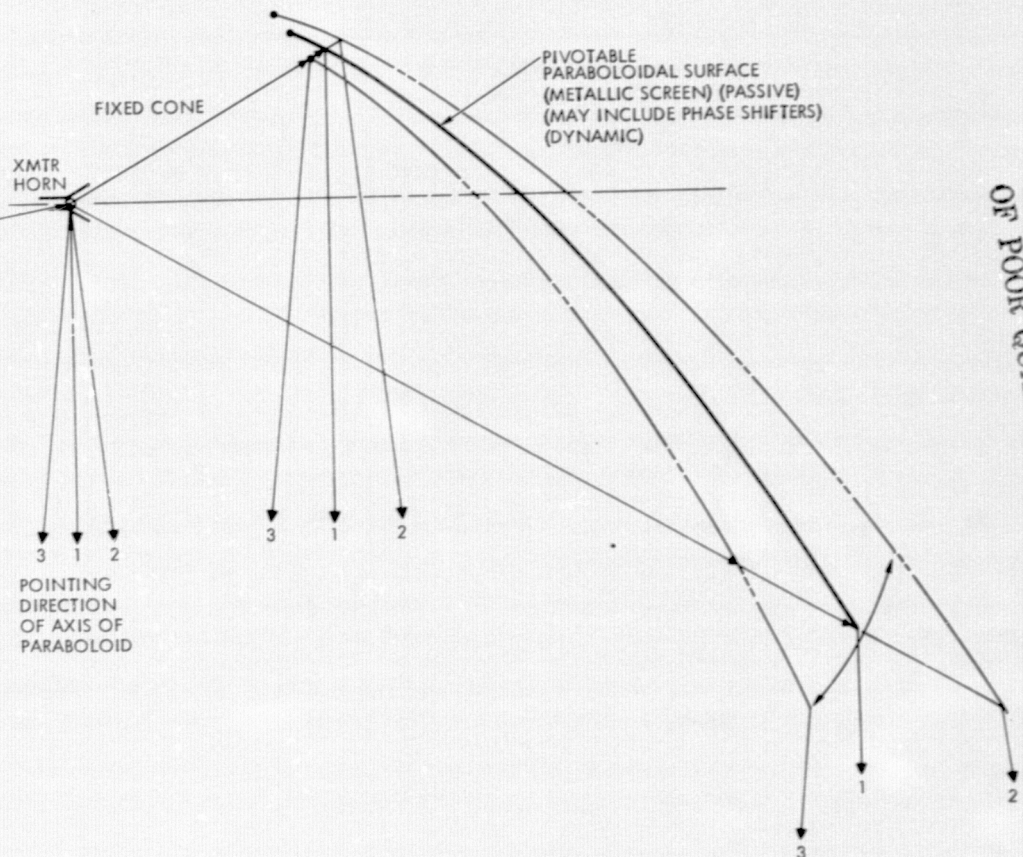
A preferred pivot angle direction is identified by rays (3) shown on the facing chart. A smaller mirror surface results. This is the reason that the antenna is located at the South Pole end of SPS for north latitude pointing and, conversely, on the North Pole end of SPS for south latitude pointing.

GEOMETRY

FLAT REFLECTOR/LENS
GEOMETRY
(FIXED XMTR HORN)



PIVOTABLE PARABOLOID
(FIXED XMTR HORN)



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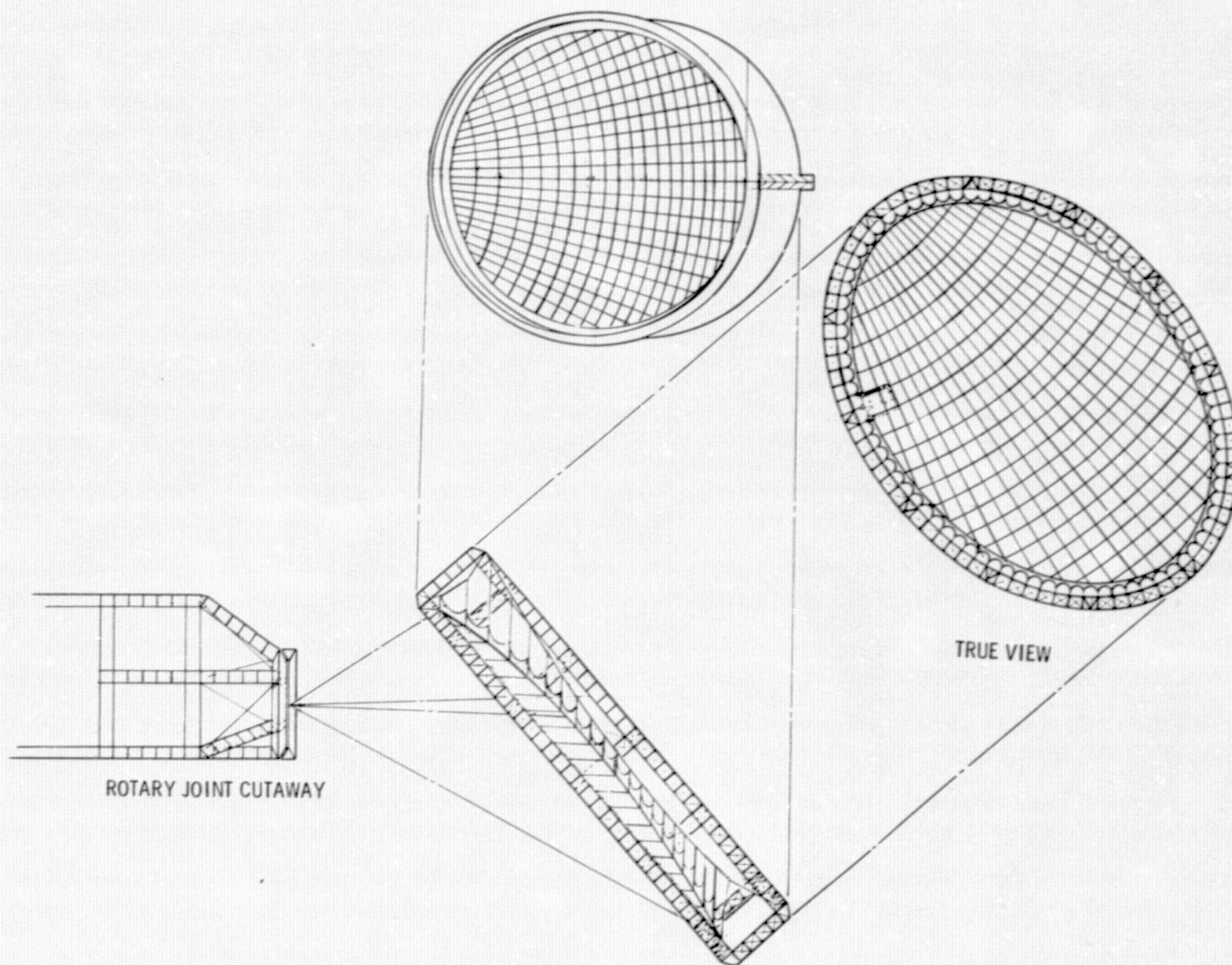
PARABOLIC ANTENNA DETAILS

The facing chart illustrates how a frame and tension wire array might provide a close tolerance wire-mesh paraboloidal mirror.

It is possible to create a paraboloidal shape if a uniform force field oriented parallel to an axis is applied to a membrane perpendicular to the axis. If a system of meridian planes radiates from the axis, each contains a flexible member (countour members) anchored to a frame—in turn, pulled upon by close uniformly spaced flexible members (force members)—and each containing the same load, a segment of a parabola is formed in each meridian plane. The exact segment of the parabola formed is determined by the angular relationship of the anchored meridian contour elements and the uniformly spaced elements representing the uniform force. Other flexible members located orthogonally to the meridian planes may be located to form concentric circles about the paraboloidal axis. The degree of surface compliance is determined by the closeness of the net pattern formed.

To effect the uniform force field upon the paraboloidal net, a system of flexible members is required with the members pulling in opposition to the previously described net. This second array of flexible members is connected to the same frame. Since only one surface (the "mirror" surface) requires shape compliance, the counter-pull member array may permit the accumulation of the uniformly spaced members (force members) into fewer major strands reacted by a single flexible member for each meridian plane formed. No concentric circular counter force element members are required. Since all flexible members are anchored on the same frame, and interconnected to each other, a complete self-contained stable system of forces and nets will be established.

PARABOLIC ANTENNA DETAILS

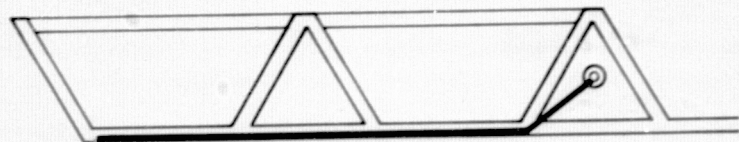
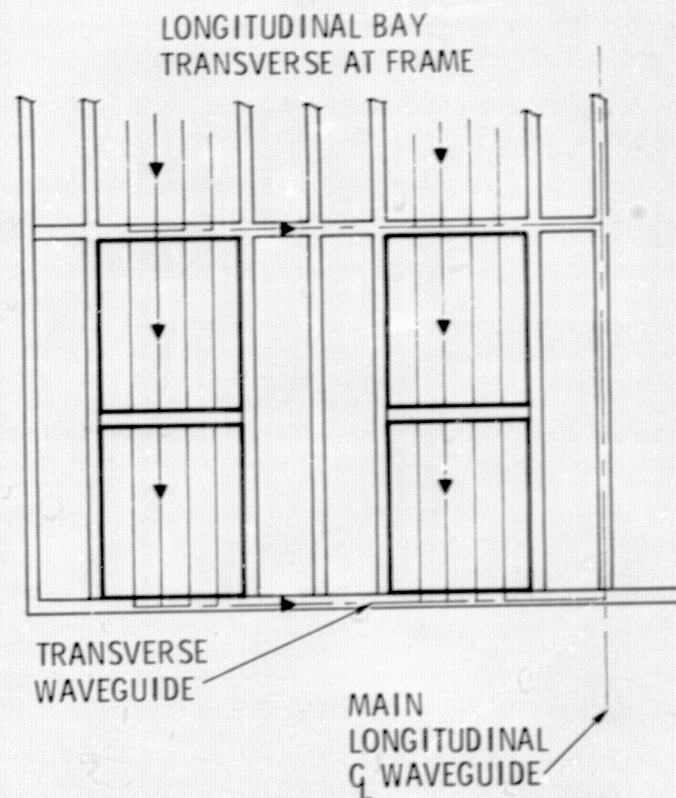
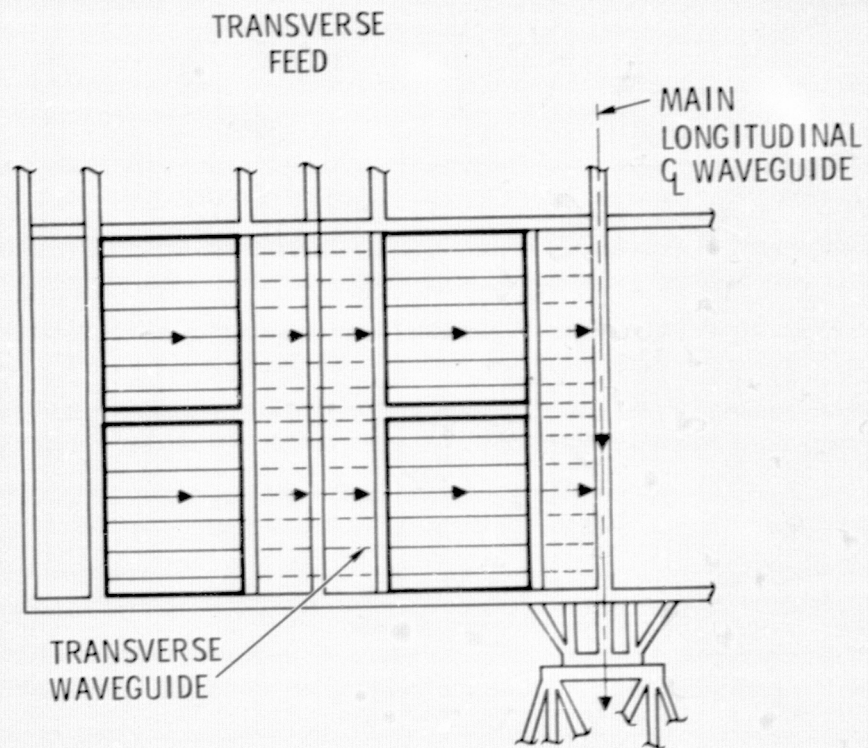


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RF WAVEGUIDE ORIENTATION OPTIONS

The facing chart is an enlargement of solar cell/waveguide options described in a previous chart. The heavy line in the lower left-hand figure illustrates the transverse waveguide relative position to adjacent supporting structure.

RF WAVEGUIDE ORIENTATION OPTIONS



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MICROWAVE—POWER—DISTRIBUTION SUBSYSTEM

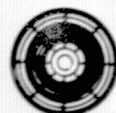
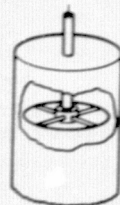
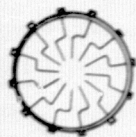
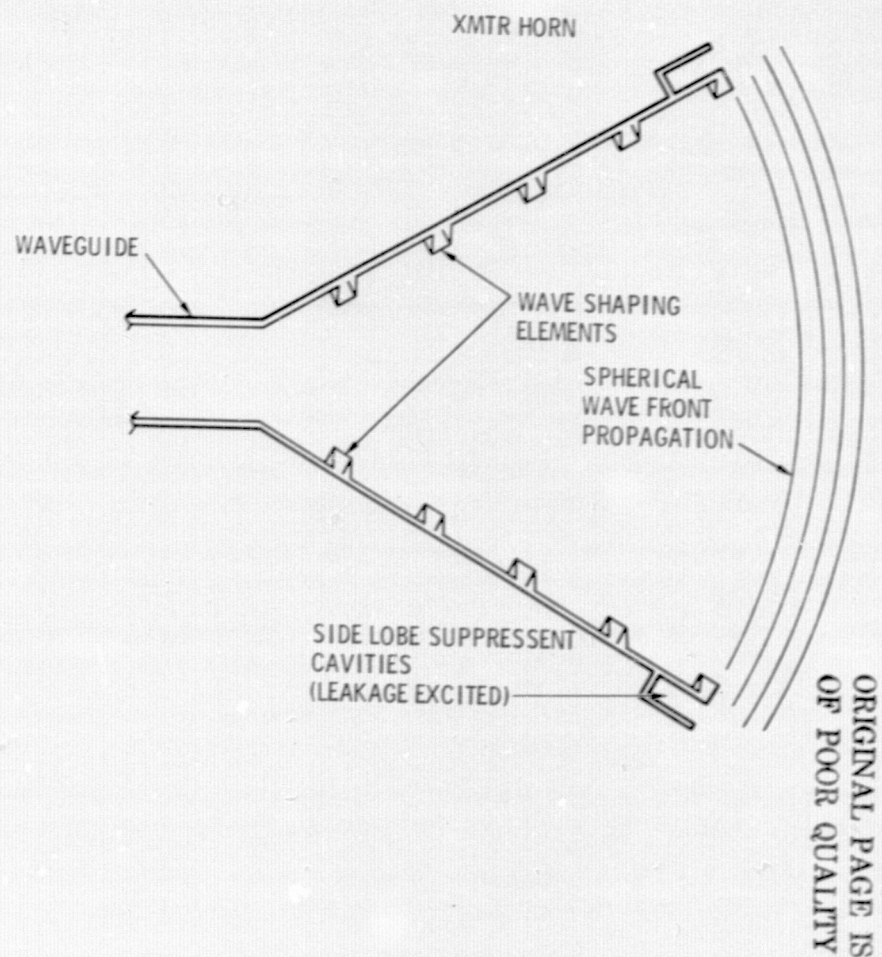
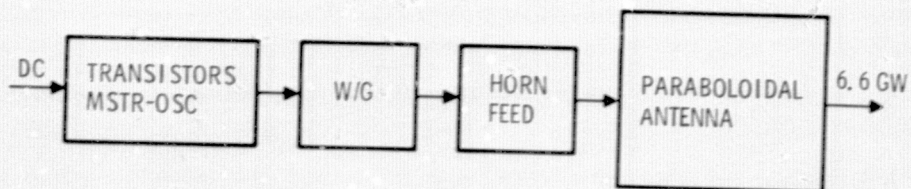
The facing chart shows:

- The solid-state circuit diagram
- Transistor mounting concepts contained in the Aerospace Corporation AIAA paper
- Transmitter horn mounted on the end of the main waveguide

All RF rotary joints described in the Aerospace Corp. AIAA paper have been eliminated. No new RF waveguide problems have been created. The lens property of the transmitter horn has been utilized on low-power (relative to SPS) radar antennas. The concept of self-excited, electrically tuneable side-lobe suppressant cavities may be new. Tuneable wave shaping of the emitted spherical wavefront may also be new. The question as to whether variable lens properties are possible must be addressed, as must the question as to whether horn spilling can be prevented and side-lobe suppression at the horn accomplished. Other questions are: What surface compliance must the paraboloidal mirror have? Will the paraboloidal mirror surface require a phase-shift property? Can the transmitter horn effectively control and phase-shift portions of the wavefront (if required) to maintain the proper power distribution at the earth rectenna? Is it possible to transmit a unique wave shape such as to produce a uniform field intensity at the rectenna? This last question poses the possibility of a 5000- to 6000-m diameter rectenna.

The attendant potential for cost reduction of the SPS system and the earth rectenna warrants a "proof-of-concept" study.

MICROWAVE-POWER-DISTRIBUTION SUBSYSTEM



(A) TRANSISTOR MOUNTING CONCEPTS

(B) COAXIAL CONDUCTOR WAVES TO TEdI WAVES CONVERTER CONCEPT

BRIEFING OUTLINE

EXECUTIVE SUMMARY

G. M. HANLEY

REFERENCE CONCEPT CHARACTERISTICS

A. A. NUSSBERGER
R. E. OGLEVIE

REFERENCE CONCEPT CONSTRUCTION

R. E. SEXTON

MICROWAVE TRANSMISSION SYSTEM

G. D. O'CLOCK

TRANSPORTATION SYSTEM

R. P. BERGERON

EXPERIMENT/ VERIFICATION PROGRAM

W. R. RHOTE



EOTV SIZING APPROACH

The EOTV power source utilizes the same construction approach as the basic SPS. Structural bays and solar blanket sizes are consistent with those of the SPS.

The payload capability of 4×10^{-6} kilograms is consistent with previous study results which indicated minimum transportation costs based on 8 to 12 EOTV flights and LEO-to-GEO trip times between 100 and 130 days.

The GaAlAs cells are assumed to be self-annealing of electron damage occurring during transit through the Van Allen Belt. A lifetime degradation in performance of 15% is consistent with basic SPS criteria. This end-of-life performance was used in all calculations.

The issue of silicon cell annealing was not addressed. However, the same assumptions used for the GaAlAs system were applied to the silicon cell configuration.

EOTV SIZING APPROACH

- SAME CONSTRUCTION/CONFIGURATION AS SPS
- PAYLOAD CAPABILITY $\geq 4 \times 10^6$ KG
- SELF ANNEALING SOLAR CELLS (GaAIAs)
- TRIP TIME LEO -TO -GEO ~ 120 DAYS
GEO -TO -LEO ≤ 30 DAYS
- END -OF -LIFE PERFORMANCE CRITERIA - 15% DEGRADATION
- SAME CRITERIA USED FOR Si EOTV CONFIGURATION



EOTV SIZING ASSUMPTIONS

The orbital parameters are consistent with SPS requirements and the delta "V" requirements, was taken from previous SEP and EOTV trajectory calculations. A 2% delta "V" margin is included in the figure given.

During occultation periods, attitude hold only is required (i.e., thrusting for orbital change is not required).

Since it is currently anticipated that thruster grid changes will be required after each mission, a minimum number of thrusters are desired to minimize operational requirements.

An excess of thrusters are included in each array to provide for potential failures and primarily to permit higher thrust from active arrays when thrusting is limited or precluded from a specific array due to thruster exhaust impingement on the solar array or to provide thrust differential as required for thrust vector/attitude control. A 5% specific impulse penalty was also applied to compensate for thrust cosine losses due to thrust vector/attitude control.

An all-electric thruster system was selected for attitude control during occultation periods. The power storage system was sized to accommodate maximum gravity gradient torques and occultation periods. A very conservative duty cycle of 100% was assumed for establishing ACS propellant requirements.

Total thrust requirements to accommodate maximum torques is estimated to be between 240 and 260 Newtons.

EOTV SIZING ASSUMPTIONS

- LEO ALTITUDE - 500 KM @ 28.5° INCLINATION
- 5700 M/SEC ΔV REQUIREMENT - INCLUDES 2% MARGIN
- SOLAR INERTIAL ATTITUDE HOLD ONLY DURING OCCULTATION PERIODS
- NUMBER OF THRUSTERS ≈ 100
- 25% SPARE THRUSTERS - FAILURES / THRUST DIFFERENTIAL
- PERFORMANCE LOSSES DURING THRUSTING - 5%
- ACS POWER REQUIREMENT - MAXIMUM OCCULTATION PERIOD
- ACS PROPELLANT REQUIREMENTS - 100% DUTY CYCLE
- ACS THRUST REQUIREMENT - 240 TO 260 NEWTONS

EOTV SIZING LOGIC

Having adopted a basic solar array configuration, the available power was thus established.

The practical upper operating temperature limit of 1900°K for molybdenum thruster grids fixes the maximum absolute operating voltage of the thrusters at 8300 volts.

An upper limit of +2000 volts was assumed in order to preclude the possibility of arcing due to LEO plasma effects. A specific trade of conductor insulation requirements as a function of positive voltage is indicated. The grid voltage establishes propellant specific impulse at 8221 sec. The number of thrusters selected establishes the remaining thruster parameters. As indicated, the number of thrusters should be selected such that the individual thrust is consistent with attitude control thrust requirements in order to preclude the need for dedicated ACS thrusters.

By establishing trip time, the maximum quantity of propellant which can be consumed during transit is established which in turn fixes maximum payload capability.

EOTV SIZING LOGIC

- SOLAR ARRAY CONFIGURATION - AVAILABLE POWER
- GRID OPERATING TEMPERATURE - MAXIMUM TOTAL VOLTAGE
- GRID VOLTAGE (PLASMA LIMITED) - SPECIFIC IMPULSE
- *NUMBER OF THRUSTERS - BEAM CURRENT/DIAMETER/THRUST
- TRIP TIME - PROPELLANT WEIGHT/PAYLOAD WEIGHT

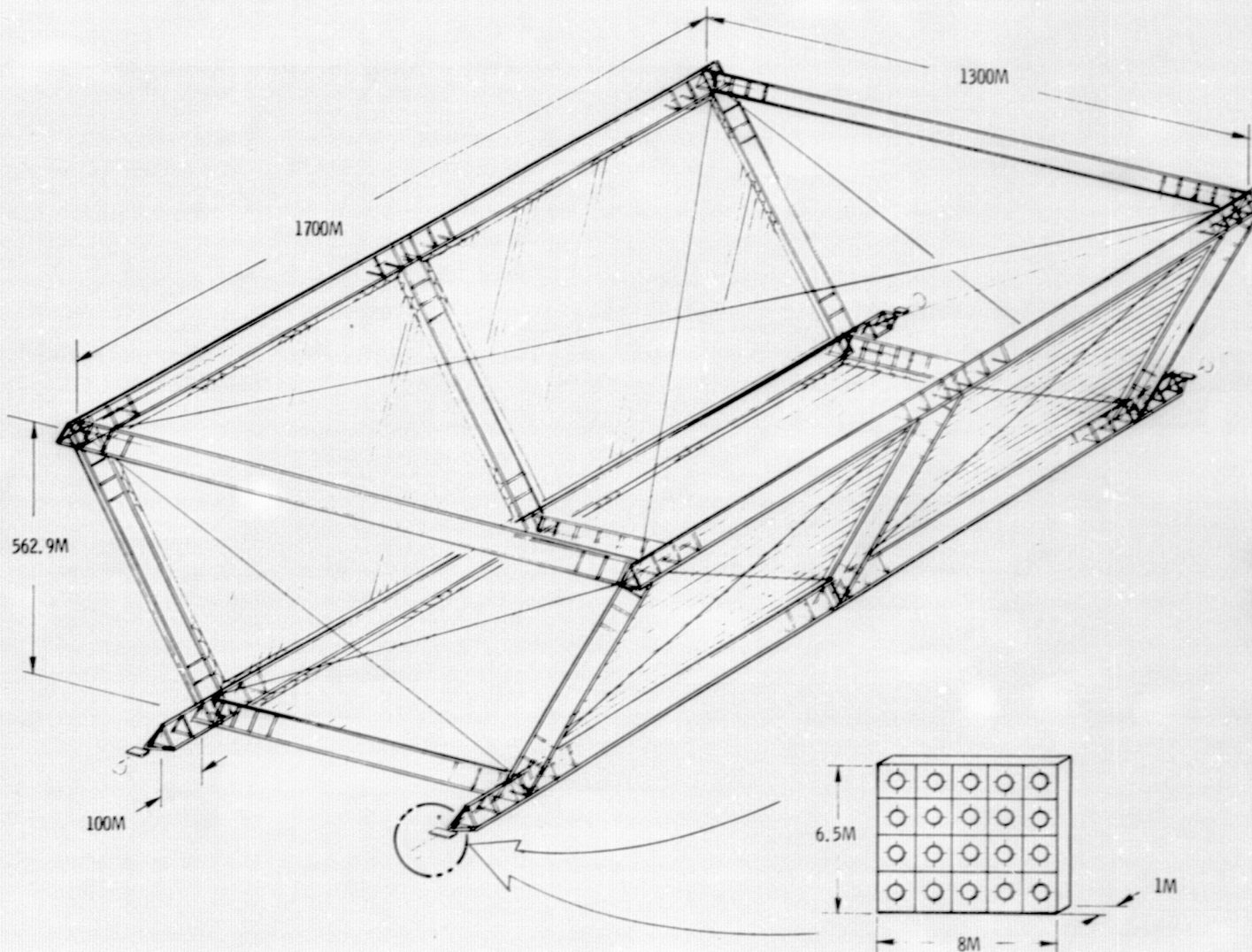
*CONSISTENT WITH ACS THRUST REQUIREMENTS



EOTV CONFIGURATION

The selected EOTV concept utilizes two SPS solar array bays, and four gimbaled thruster module arrays. Each thruster array contains twenty thrusters, however, only sixty-four thrusters are capable of firing simultaneously. Four electric thrusters utilizing stored electrical power are used for attitude hold only during occultation periods.

EOTV CONFIGURATION



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EOTV SOLAR ARRAY AND POWER CHARACTERISTICS

The solar array consists of two SPS bays with a total power output of 335.5 megawatts. Line losses of 6% and an end-of-life cell degradation of 15% were assumed which yields a net power to the thruster arrays of 268.1 megawatts. The thruster array losses were determined to be negligible.

The power storage system was sized on the same basis as for the SPS, 200 kilowatt-hours per kilogram weight.

EOTV SOLAR ARRAY AND POWER CHARACTERISTICS (GaAIAs)

- TWO SPS BAY SOLAR CELL ARRAY
- TOTAL POWER AT CELLS - 335.5 MEGAWATTS
- LINE LOSSES - 6%
- CELL DEGRADATION - 15%
- NET POWER TO THRUSTERS - 268.1 MEGAWATTS
- POWER STORAGE - 200 KWH/KG



EOTV THRUSTER CHARACTERISTICS

Primary thruster characteristics are presented. The number of thrusters were arrived at as previously described. The thrusters are mounted in four arrays of 20 each. Sixty-four thrusters only are operating simultaneously to provide orbital transfer thrust and a total of four are required to maintain attitude control during occultation periods.

EOTV THRUSTER CHARACTERISTICS

- MAXIMUM OPERATING TEMPERATURE - 1900⁰ K
- TOTAL VOLTAGE - 8300 VOLTS
- GRID VOLTAGE - 2000 VOLTS MAXIMUM
- BEAM CURRENT - 1904 AMP
- SPECIFIC IMPULSE - 8221 SEC
- THRUSTER DIAMETER ~ 76 CM
- THRUST/THRUSTER - 69.7 NEWTON
- NUMBER OF THRUSTERS - 80 (INCLUDES 25% SPARES)



EOTV WEIGHT SUMMARY

This table is self-explanatory. The same growth margin as used for the SPS has been applied.

EOTV WEIGHT SUMMARY (KG)
(GaAlAs)

SOLAR ARRAY		493,056
CELLS/STRUCTURE	263,510	
POWER CONDITIONING	229,546	
THRUSTER ARRAY (4)		104,046
THRUSTERS/STRUCTURE	6,152	
CONDUCTORS	270	
BEAMS/GIMBALS	2,256	
PROPELLANT TANKS	95,368	
ATTITUDE CONTROL SYSTEM		50,471
POWER SUPPLY	48,258	
SYSTEM COMPONENTS	1,000	
PROPELLANT TANKS	1,213	
EOTV INERT WEIGHT		647,573
25% GROWTH		161,893
TOTAL INERT WEIGHT		809,466
PROPELLANT WEIGHT		547,294
EOTV LOADED WEIGHT		1,356,760
PAYLOAD WEIGHT		5,310,568
LEO DEPARTURE WEIGHT		6,667,328



EOTV PROPELLANT SUMMARY

This table is self explanatory. The GEO-to-LEO transit time is based on all sixty-four thrusters operating.

EOTV PROPELLANT SUMMARY (KG) (GaAlAs)

DELTA V PROPELLANT

540,420

LEO -TO -GEO

478,170

GEO -TO -LEO

62,250

ACS PROPELLANT (4 THRUSTERS)

6,874

LEO -TO -GEO

5,977

GEO -TO -LEO

897

TOTAL PROPELLANT WEIGHT

547,294

NOTE:

120 DAY TRIP TIME LEO-TO-GEO (100 DAY THRUST TIME)
16 DAY TRIP TIME GEO-TO-LEO (13 DAY THRUST TIME)



GaAlAs AND SILICON POWERED EOTV WEIGHT COMPARISON

This table is self-explanatory. The primary difference in configurations is in the solar array weights which results in a 14% reduction in payload capability for the silicon powered EOTV.

GaAIAs AND SILICON POWERED EOTV WEIGHT COMPARISON (KG)

<u>ELEMENT</u>	<u>GaAIAs</u>	<u>SILICON</u>
SOLAR ARRAY	493,056	1,032,991
THRUSTER ARRAY	104,046	113,355
ATTITUDE CONTROL SYSTEM	50,471	50,576
EOTV INERT WEIGHT	647,573	1,196,922
GROWTH - 25%	161,893	299,231
TOTAL EOTV INERT WT.	809,466	1,496,153
DELTA V PROPELLANT	540,420	593,170
ACS PROPELLANT	6,874	7,471
TOTAL EOTV LOADED WT.	1,356,760	2,096,794
PAYLOAD WEIGHT	5,310,568	4,570,534
LEO DEPARTURE WT.	6,667,328	6,667,328
TRIP TIME (UP/DOWN)	120/16	120/28



CONFIGURATION TRADE OPTIONS

The effects of lowering solar array voltage from the baseline of -8300 volts were evaluated. As indicated, the effects are negligible. Although the thruster array weight increases by 2.5 times, the overall impact on EOTV weight is not significant. The added weight of the thruster arrays could be offset by reduction in power conditioning and conductor insulation weights.

The effect of reducing the number of thrusters was also evaluated and, as indicated, the only effect of any consequence would be the addition of dedicated ACS thrusters as the ACS thrust requirement is only 50% of the available thrust.

CONFIGURATION TRADE OPTIONS

- REDUCE TOTAL ABSOLUTE VOLTAGE REQUIREMENT - 5500 VOLTS

MAINTAIN + 2000 VOLT GRID VOLTAGE

THRUSTER DIAMETER INCREASES TO ~ 120 CM

GRID TEMPERATURE DECREASES TO $\sim 1500^{\circ}\text{K}$

THRUSTER ARRAY WEIGHT INCREASES ~ 2.5 TIMES

ADVANTAGEOUS ONLY IF POWER CONDITIONING WEIGHT DECREASES

- REDUCE NUMBER OF THRUSTERS - $1/2$

BEAM CURRENT AND UNIT THRUST DOUBLES

THRUSTER DIAMETER INCREASES TO ~ 108 CM

THRUSTER ARRAY WEIGHT - NO SIGNIFICANT CHANGE

DEDICATED ACS THRUSTERS REQUIRED



EOTV SUMMARY

The EOTV configuration presented does not necessarily reflect cost optimum configuration. Further trade studies are indicated in the areas of quantity of thrusters, payload size and trip time as related to other elements of program cost.

The depth of configuration definition is consistent with contract resources available and is believed to be adequate for overall SPS program requirements assessment.

Since thruster performance is directly proportional to grid voltage, further assessment of near earth plasma effects are required. Thruster performance data is consistent with electric ion propulsion technology as obtained from NASA-LeRC.

EOTV SUMMARY

- CONFIGURATION NOT NECESSARILY OPTIMUM

NUMBER THRUSTERS

PAYLOAD SIZE

TRIP TIME ASSESSMENT

- CONFIGURATION DEFINITION

CONSISTENT WITH AVAILABLE RESOURCES

ADEQUATE FOR CURRENT SPS REQUIREMENTS DEFINITION

- THRUSTER PERFORMANCE

GRID VOLTAGE LIMITATION - NEAR EARTH PLASMA EFFECTS

MAY BE VERIFIED BY NASA-LeRC

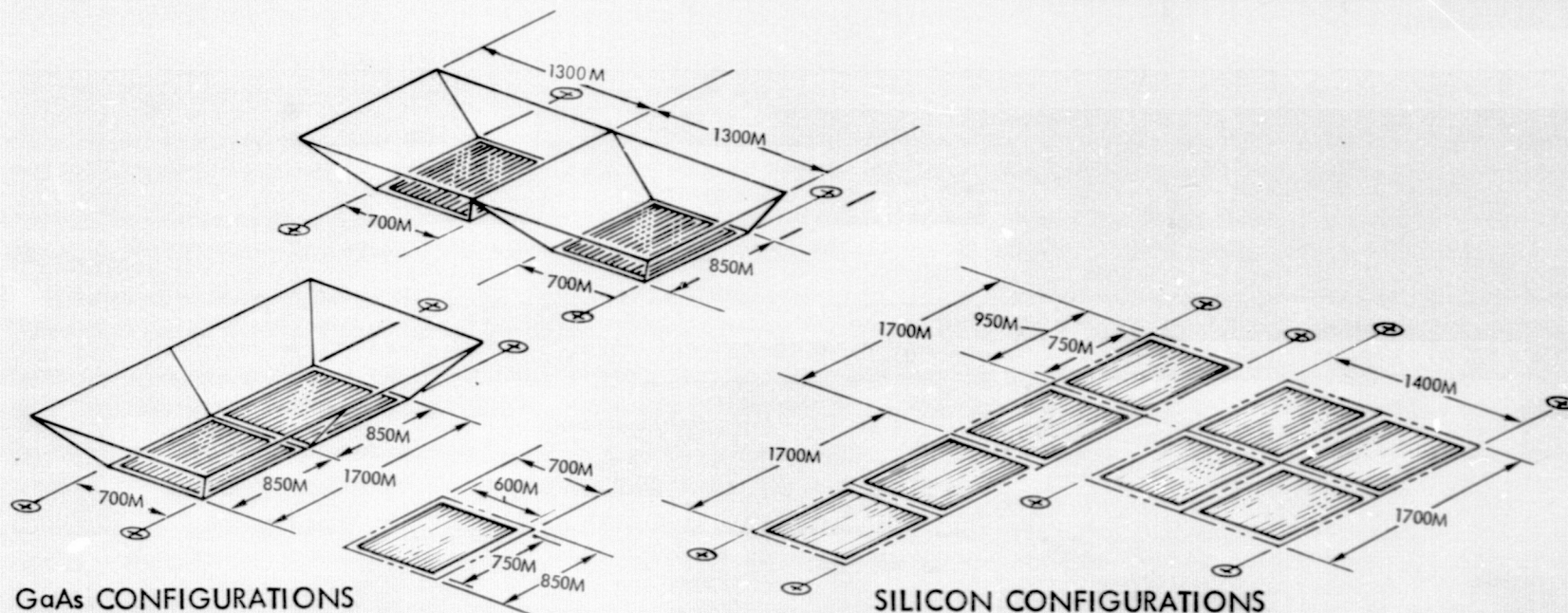
- SILICON CELL ANNEALING NOT ADDRESSED IN STUDY



EOTV SOLAR ARRAY COMPARISONS (GaAs VS SI SOLAR CELLS)

A comparison was made of the EOTV requirements using GaAs and Silicon solar cells. The configurations used in the comparison are shown with a tabulation of solar array parameters and values. The silicon solar array weights are 725,904 kg compared to 263,511 kg driven by higher specific weight ($.426 \text{ kg/m}^2$ vs $.252 \text{ kg/m}^2$) and requirement for larger area ($1,704,242 \text{ m}^2$ vs $886,950 \text{ m}^2$). The impact of reflector weight on the GaAs configuration is negligible.

EOTV SOLAR ARRAY CONFIGURATIONS (GaAs VS Si SOLAR CELLS)



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GaAs CONFIGURATIONS

SILICON CONFIGURATIONS

NOTE:

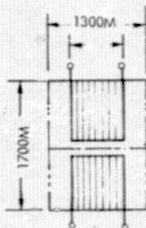
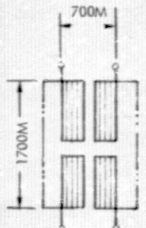
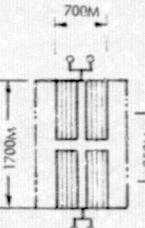
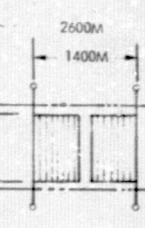
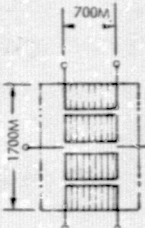
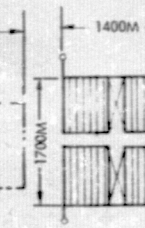
(1) NO SPACE DEGRADATION
ALLOWANCES

PARAMETER	GaAs	SILICON
SOLAR INPUT	1319.5 W/M ²	1319.5 W/M ²
ENERGY ONTO CELLS	2414.7 (CR = 1.83)	1319.5 (CR = 1)
η (T)	424.98 (17.6%)	221.17 (16.74%)
DESIGN FACTOR	278.24 (.89)	196.85 (.89)
POWER OUTPUT (ARRAY) (1)	335.48 MEGAWATTS	335.48 MEGAWATTS
AREA REQM'T	886,950 M ²	1,704,242 M ²
ARRAY AREA	900,000 M ²	1,800,000 M ²
ARRAY WEIGHT (KG)	223,511 (.252 KG/M ²)	725,904 (.426 KG/M ²)
REFLECTOR AREA	2,210,000 M ²	-
REFLECTOR WEIGHT	40,000 KG	-
SUBTOTAL	263,511 KG	725,904 KG

EOTV POWER DISTRIBUTION AND CONTROL WEIGHT COMPARISONS

This chart shows the power distribution and control weight comparisons for the different EOTV configurations studied. A solar array voltage output of 2080 V dc was selected as the upper limit for power generation to stay within tolerable plasma power losses for low earth orbit operations. At 2080 V dc power distribution the lowest weight concept results in a power distribution subsystem weight of 288,440 kg. This configuration is a direct energy transfer to the engines. This weight was calculated at a distribution (line loss) efficiency of 94% (i.e., 6% line loss). The weight calculations ranged up to 632,900 kg dependent upon specific configuration details. A negative voltage system was compared to show impact of higher voltage. A negative 6300 volts was selected for this purpose since this is the second voltage requirement of the EOTV thruster system. This concept requires power conditioning at the thrusters to provide the +2000 volt inputs required. The silicon system was compared for the lowest weight approach and results in a weight penalty of ~33% (307,090 kg vs 229,550 kg). The +2080 volt concept is the recommended approach since it does not require major power conditioning (i.e., direct power transfer) and the -6300 volt system is susceptible to arcing problems in the plasma environment.

EOTV POWER DISTRIBUTION AND CONTROL WEIGHT COMPARISONS

EOTV CONFIGURATION							
CELL MAT'L CR TRANS. VOLTAGE PANEL CONFIG.	GaAs 2 +2080V SPLIT 2 PANELS	SPLIT 4 PANELS	SPLIT 4 PANELS	SPLIT 2 PANELS	SPLIT 4 PANELS	-6300V SPLIT 2 PANELS	SILICON 1 -6300V SPLIT 4 PANELS
WEIGHTS (10 ⁶ KG)							
INTERTIES	221,940	67,260	177,550	177,550	177,500	19,540	19,540
MAIN FEEDERS	144,520	119,230	57,810	57,810	57,810	22,850	83,740
SUMMING BUS	177,550	44,390	177,550	177,550	55,490	68,800	68,800
TIE BARS	24,660	24,660	24,660	24,660	24,660	8,140	8,140
SW GEARS	2,290	2,290	2,290	2,290	2,290	9,460	7,310
POWER CONDIT.	-	-	-	-	-	75,490	75,490
INSUL.	4,400	4,400	4,400	4,400	4,400	4,400	16,150
SEC. STRUCT.	57,540	26,220	44,200	44,430	3,220	20,870	27,920
TOTAL	632,900	288,440	486,180	488,690	354,420	229,550	307,090

NOTE: CORRECTION FACTORS

TRANS. EFF.	92%	94%	96%
FACTOR	.756	1.0	1.319

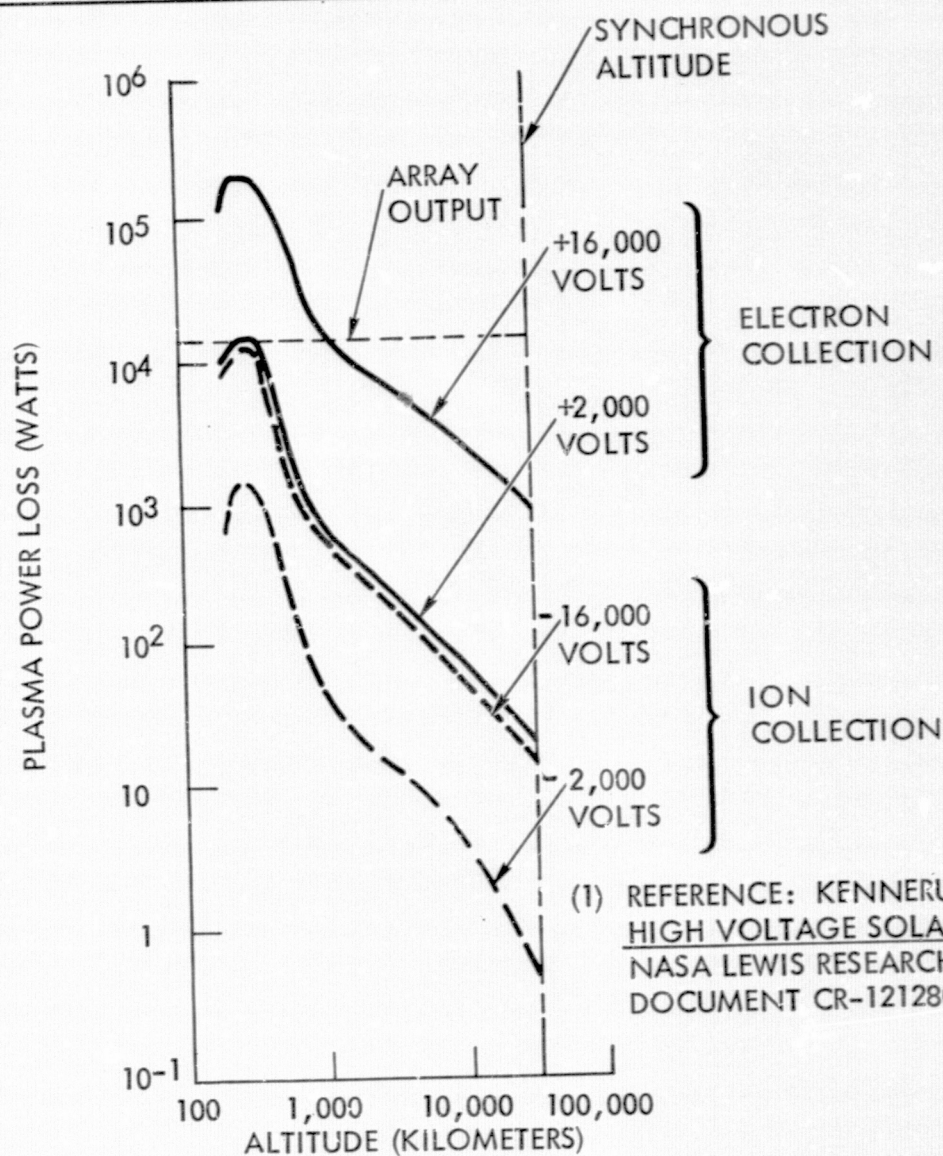


PLASMA POWER LOSSES FROM A 15 KW SOLAR
ARRAY WITH 90% INSULATING SURFACE

The solar array voltages must be as high as possible to reduce wiring weight penalties, yet, power loss by current leaking through the surrounding plasma must be at an acceptable level. There is no significant flight test data available on plasma-current leakage. Planned experiments aboard the SPHINX satellite (February 1974) were lost due to a launch failure. K. L. Kennerud in 1974 predicted plasma power loss based on analysis and plasma-chamber experiments. The plasma loss from a 90 percent insulated array is plotted in the figure as a function of altitude with voltage as a parameter. At 500 km altitude and very large arrays and high efficiency cells, it may be possible to utilize 2000 volts. Considerable more data is required to determine the upper limit of usable voltage.

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PLASMA POWER LOSSES FROM A 15 KW SOLAR ARRAY WITH 90% INSULATING SURFACE⁽¹⁾



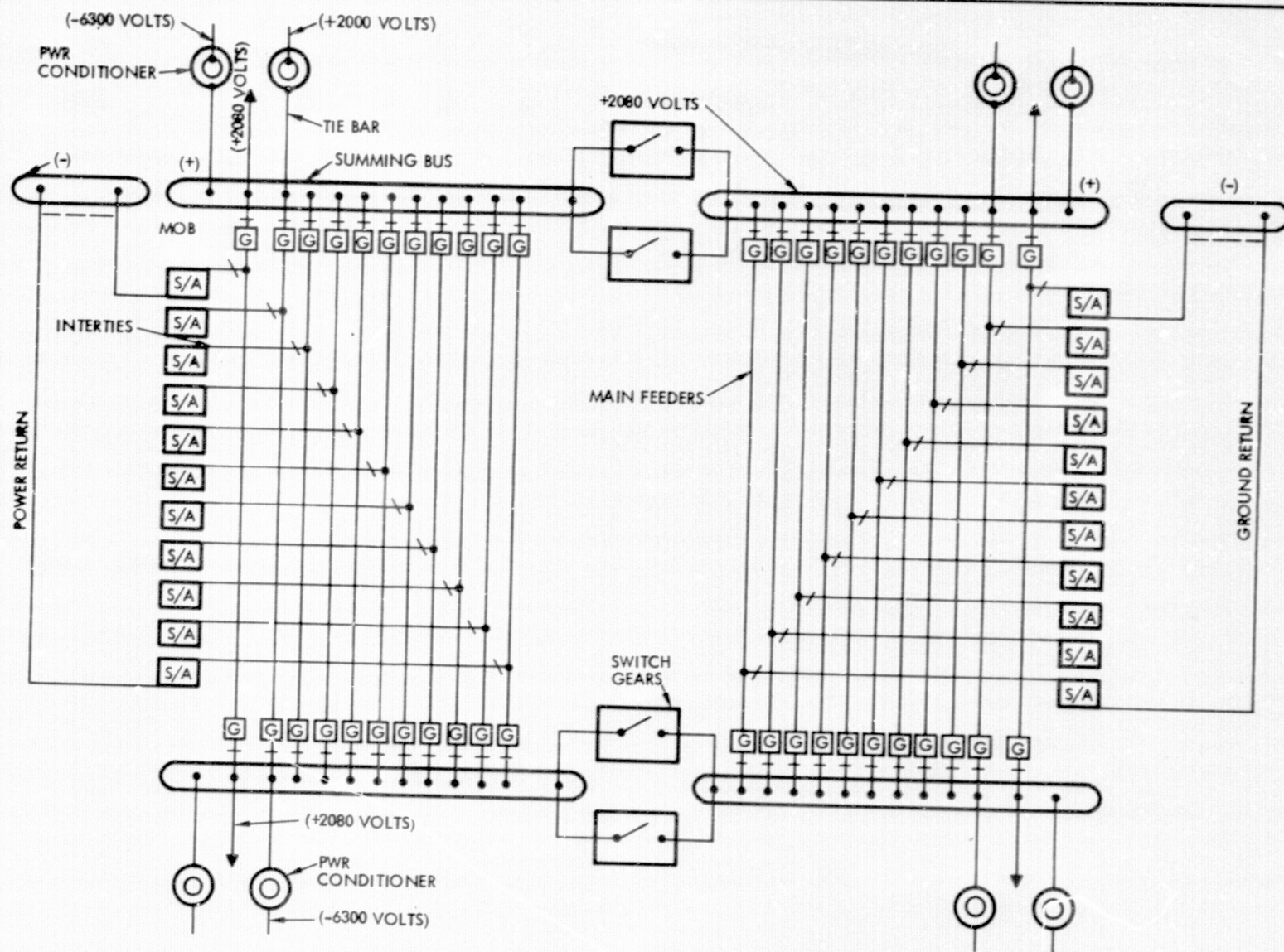
(1) REFERENCE: KENNERUD, K.L.
HIGH VOLTAGE SOLAR ARRAY EXPERIMENTS.
NASA LEWIS RESEARCH CENTER
DOCUMENT CR-121280 1974



EOTV POWER DISTRIBUTION SIMPLIFIED BLOCK DIAGRAM

This chart illustrates the EOTV power distribution interface for the solar photovoltaic concept. The distribution subsystem consists of interties, main feeders, summing bus, tie bar, switch gears, and dc/dc converters. The solar arrays feed the load buses with a direct energy transfer. Provisions are included to switch power from any bus to any thruster location. The basic voltages supplied are +2000 V dc and -6300 V dc. Individual power supplies will be included as required at the thrusters to supply other voltages.

EOTV POWER DISTRIBUTION SIMPLIFIED BLOCK DIAGRAM



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BRIEFING OUTLINE

EXECUTIVE SUMMARY

G. M. HANLEY

REFERENCE CONCEPT CHARACTERISTICS

A. A. NUSSBERGER
R. E. OGLEVIE

REFERENCE CONCEPT CONSTRUCTION

R. E. SEXTON

MICROWAVE TRANSMISSION SYSTEM

G. D. O'CLOCK

TRANSPORTATION SYSTEM

R. P. BERGERON

EXPERIMENT/ VERIFICATION PROGRAM

W. R. RHOTE



SPS TECHNOLOGY VERIFICATION

- SPS DEVELOPMENT PLANNING SCENARIO OVERVIEW
- EVOLUTION OF SPS GEOSTATIONARY TEST PLATFORM
- POWER AMPLIFIER DEVELOPMENT TREND
- MICROWAVE POWER TECHNOLOGY VERIFICATION
GOALS, OBJECTIVES AND PLANS
- EXPLORATORY RESEARCH PLAN FOR CRITICAL MPTS
COMPONENT DEVELOPMENT



SPS DEVELOPMENT PLANNING SCENARIO OVERVIEW - 1978

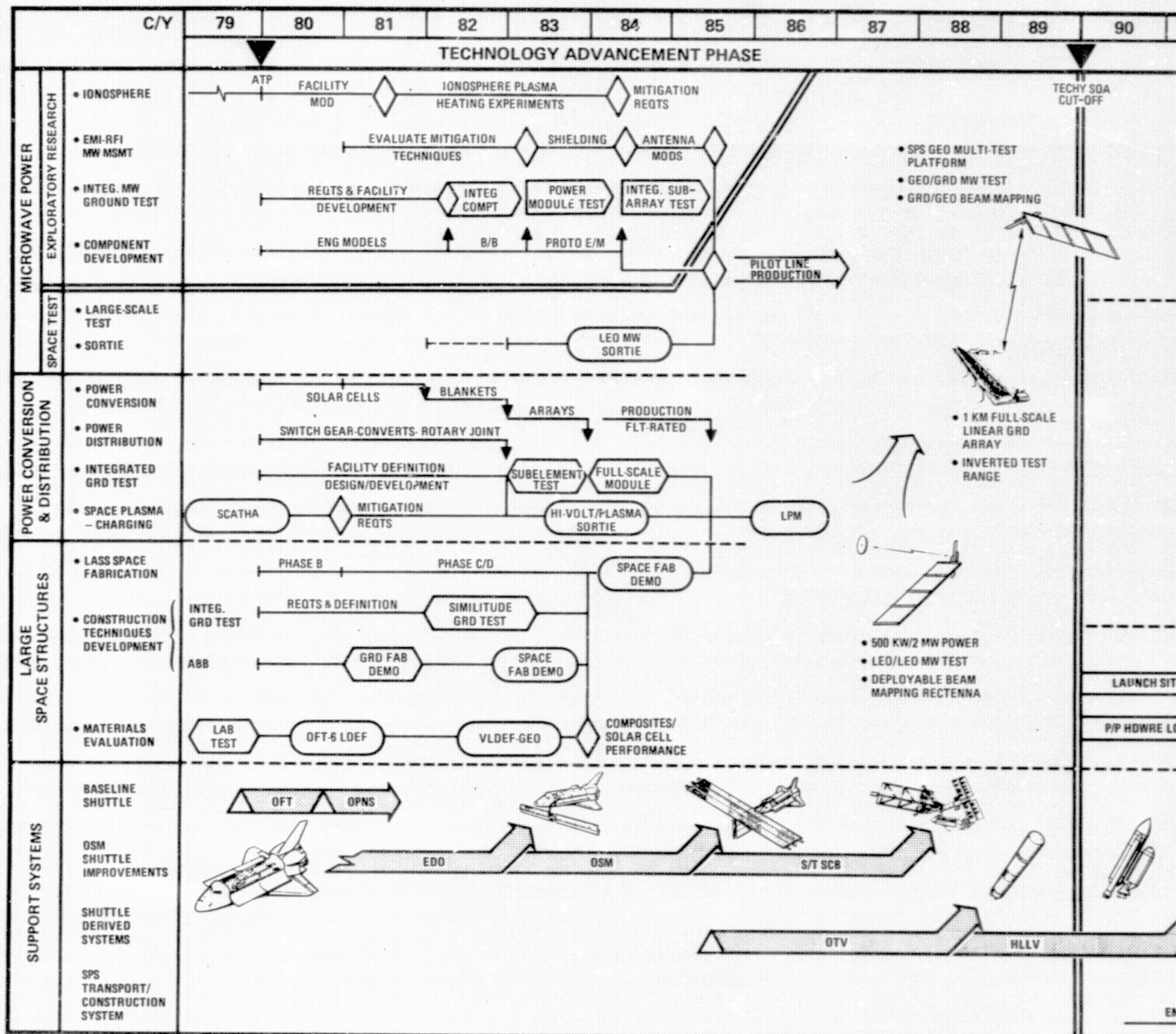
An updated SPS development planning scenario overview was developed during this period to reflect a perceived consensus of current development planning options. The scenario is pictorialized as a synthesized program reflecting current recommendations and projected 5 year plans for exploratory research, technology advancement and pilot-plant demonstration.

The exploratory research plan segment is contained within the double lines and provides the "seed-bed" for prototype MPTS development. Planned and on-going NASA enabling technology for power conversion/distribution and large space structures development are structured to provide evolutionary timelines leading to large SPS-type subscale space test articles during the end of the decade. The commitment to large-scale SPS development ground and space test activity will come about 1985 based upon Exploratory Research Program results and will require funding levels on the order of several billions of dollars.

The pilot-plant demonstration phase reflects in general the "precursor" demonstration concept proposed as the most cost-effective approach to large scale pilot-plant commercial end-to-end performance demonstration. Utilization of Shuttle-derived low-cost launch vehicles with LEO assembly and dedicated electric propulsion orbital transfer to geosynchronous is baselined.

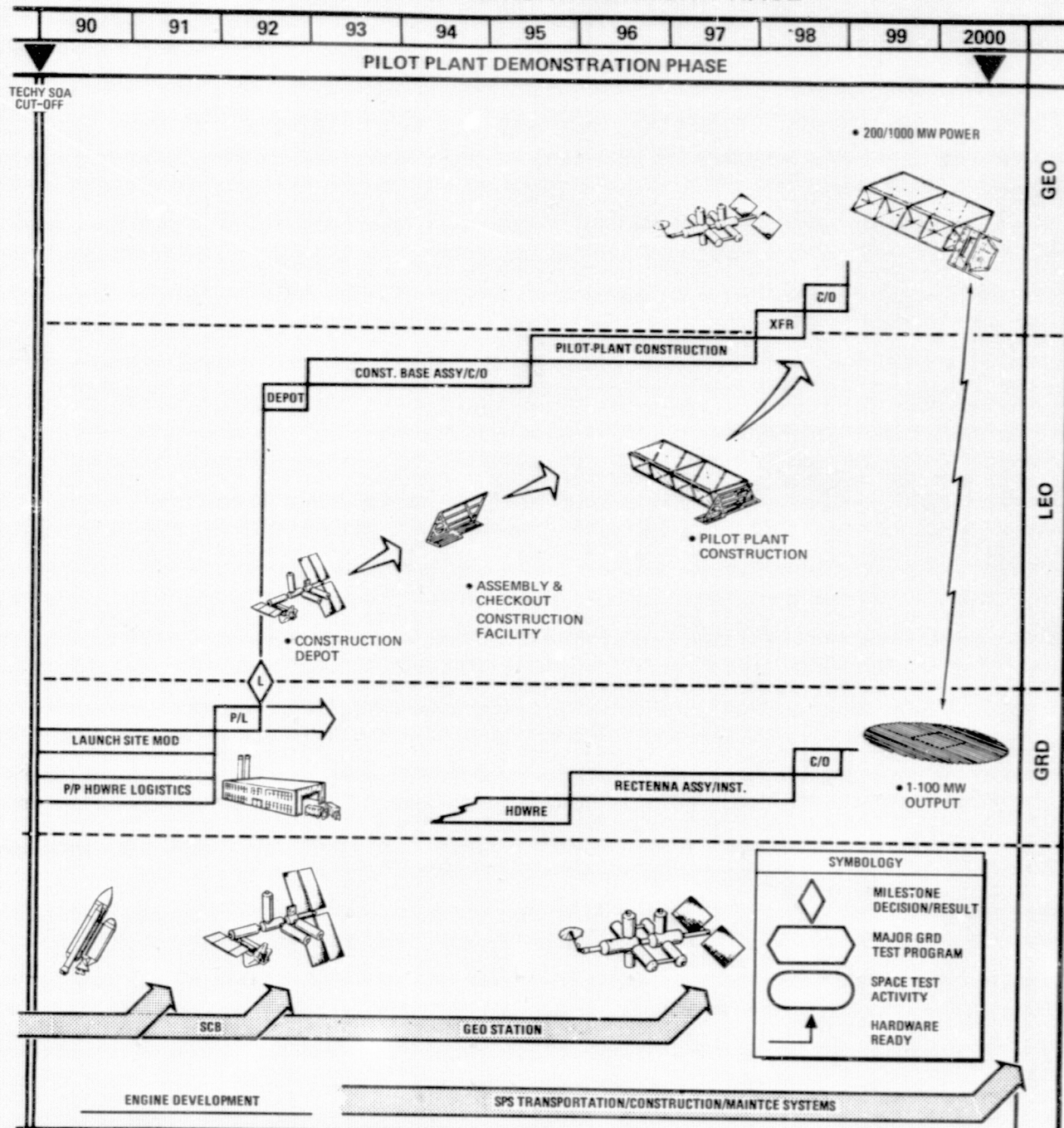
The pilot plant demonstration platform configuration, while driven primarily by SPS prototypical design parameters, will also provide the Nation with a long-life, multi-megawatt level power station in geosynchronous orbit which could be categorized as a National Geostationary Space Facility with capability and modular provisions for communications/navigation antenna farms and extended manned earth surveillance capabilities on a self-sustaining basis. The benefits provided by these multi-mission services would add considerable cost-effectiveness to the SPS pilot plant concept.

SPS TECHNOLOGY ADVANCEMENT PLAN



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SPS PILOT PLANT DEMONSTRATION PHASE



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PILOT PLANT DEMONSTRATOR DEVELOPMENT PHILOSOPHY

A primary guideline for SPS pilot plant development involves the utilization of the EOTV as the power module. This allows common development and construction utilizing a single modular construction facility that can be expanded into the basic SPS assembly fixture.

An evolutionary construction scenario is postulated wherein the basic construction facility is fabricated in LEO and utilized in low orbit to construct first the pilot plant and then the operational EOTV's. The construction facility would then be upgraded to SPS production capability and transferred to GEO by EOTV.

The pilot plant demonstrator would be sized to the projected EOTV power level of 335 MW at the array. Allowing for radiation degradation and power distribution losses, power to the microwave antenna would be approximately 285 MW. Microwave transmission losses would reduce this value to about 230 MW at the rectenna. This would result in recovery of 8 MW of power for a 7 km diameter rectenna or 2 MW of power for a 1.75 km rectenna.

PILOT PLANT DEMONSTRATOR DEVELOPMENT PHILOSOPHY

- EOTV - PILOT PLANT - PROTOTYPE CONSTRUCTION COMMONALITY
- COMMON EVOLVING CONSTRUCTION FACILITY
 - ✓ LEO CONSTRUCTION OF EOTV AND PILOT PLANT
 - ✓ GEO CONSTRUCTION OF SPS
- PILOT PLANT SIZED TO EOTV POWER LEVEL
- FIRST UNIT IS PILOT PLANT - EOTV + MW ANTENNA
- CONSTRUCTION FACILITY
 - ✓ FABRICATED IN LEO
 - ✓ REMAINS IN LEO TO CONSTRUCT PILOT PLANT AND EOTV'S
 - ✓ TRANSFERRED BY EOTV TO GEO FOR SPS CONSTRUCTION



SPS LEO/GEO TEST PLATFORM EVOLUTION

An essential requirement exists for major SPS subsystem test and evolution at LEO and geosynchronous altitude late in the 80's. Current NASA planning projects a major SPS subscale test article for LEO to LEO microwave testing. This same test platform can be boosted via electric propulsion to GEO to serve as an SPS multi-test platform in both a GEO to Ground and Ground to GEO mode (functioning as an inverted test range with the addition of pilot beam electronics).

It appears likely that SPS orbital test requirements and test system definition will have to be satisfied through the evolutionary utilization of NASA large structure space projects that evolve and are funded. Requirements for multi-mission applicability, construction modularity, and a broad range of power, size, and mass options which include SPS test verification objectives will need to be evaluated and synthesized. The degree of SPS technical issue resolution will be a function of the commonality between LSS/LPM evolving developments and the final SPS system definition characteristics.

Two perceived large structure development thrusts that might evolve into an SPS test platform configuration are the Large Space Structures Program and the SEP/PEP/LPM power platform focused effort. The issue of GaAs versus Silicon solar cells for the SPS baseline configuration is a key determinant in the usefulness of projected power module developments as SPS space test platforms.

SPS LEO/GEO TEST PLATFORM EVOLUTION

- LARGE SCALE SPS ORBITAL TEST PLATFORM WILL MOST LIKELY EVOLVE AS AN EXTENSION OF PLANNED LSS/LARGE POWER MODULE DEVELOPMENTS
- DEGREE OF SPS KEY TECHNOLOGY ISSUE RESOLUTION IS A FUNCTION OF THE COMMONALITY BETWEEN LSS/LPM DEVELOPMENT AND FINAL SPS DESIGN PARAMETERS
- ECONOMIC AND LEAD-TIME CONSIDERATIONS DICTATE ONE SPS TEST PLATFORM FOR BOTH LEO AND GEO TESTING
- TWO PERCEIVED EVOLUTIONARY DEVELOPMENT PATHS
 - ✓ LSS/SPACE CONSTRUCTION THRUST (BEAM MACHINE)
 - ✓ SEP/PEP/LPM THRUST (DEPLOYABLE/ERECTABLE)
- THE SOLAR CELL ISSUE IS CRITICAL (GaAs VERSUS SILICON)



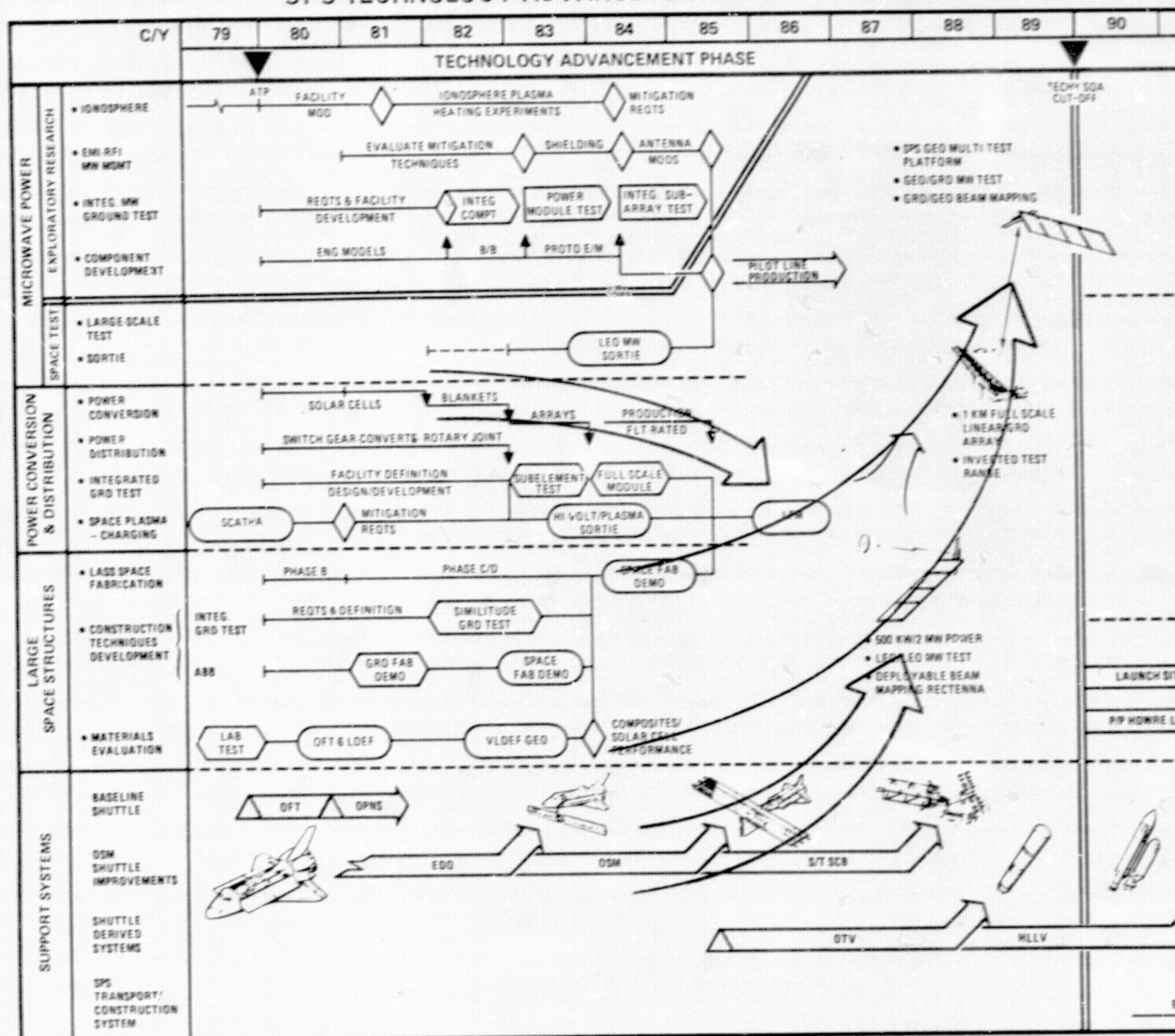
SPS TEST PLATFORM DEVELOPMENT THRUSTS

The current SPS Technology Advancement Plan baselines a large power module development (250/500 kW) by 1986 which can be modified as an SPS microwave transmission test article for use both at LEO and GEO.

The configuration of this large power system can be considered indeterminate at this time and will evolve as a product of the large space structures technology program and/or SEPS/PEP/OSM power module evolutionary development. With respect to the extent to which these development options satisfy initial SPS technology verification objectives, the issues involve:

- Beam machine space fabrication versus deployable/erectable construction
- Silicon solar arrays versus GaAs solar arrays versus hybrid array combinations
- Capability for both LEO and GEO test operations

SPS TECHNOLOGY ADVANCEMENT PLAN



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SPS DEMONSTRATION TEST ARTICLE IMPLICATIONS

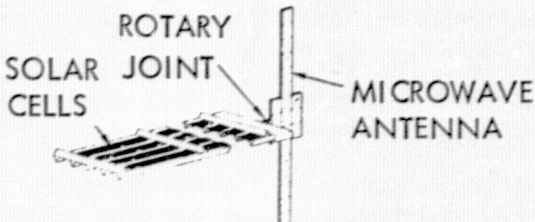
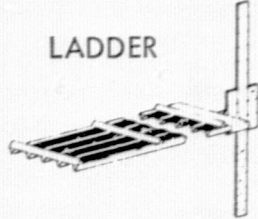
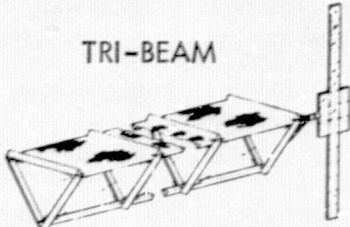
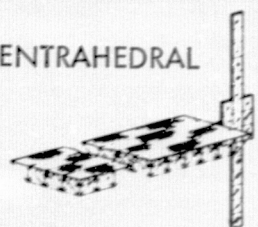
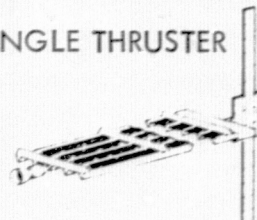
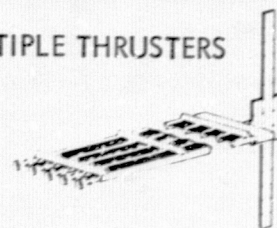
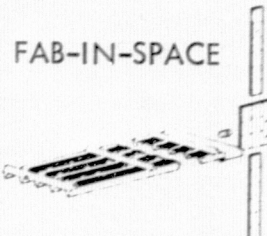
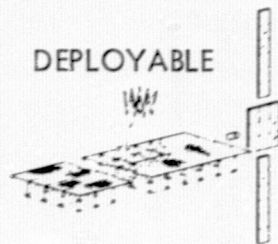
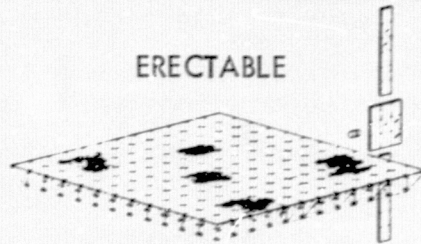
The configuration of the projected SPS development flight test article is influenced by structural, construction and propulsion system options as shown on the facing chart. Each of these potential influences must be evaluated as part of large space structures construction analysis when determining possible concept variations.

Structural variations for both the solar array support and the microwave power antenna may dictate different construction assembly fixtures and techniques. The ladder, tribeam and pentahedral truss structural configurations have differing implications and merits in terms of project application, SPS legacy and types of construction.

The requirement to relocate the SPS test article to a higher orbit will necessitate provisions for orbital transfer propulsion units which can significantly influence the system design.

Variations in the method of assembly - whether erectable, deployable or fabricated in space will drive the construction system.

SPS DEMONSTRATION TEST ARTICLE IMPLICATIONS

BASIC CONFIGURATION	 Diagram illustrating the basic configuration of a Space Power System (SPS) test article. It shows a central vertical structure with a rotary joint connecting to a horizontal arm. The arm carries solar cells and a microwave antenna.		
STRUCTURAL IMPLICATIONS	 LADDER	 TRI-BEAM	 PENTRAHEDRAL
PROPULSION IMPLICATIONS	 SINGLE THRUSTER	 MULTIPLE THRUSTERS	
CONSTRUCTION IMPLICATIONS	 FAB-IN-SPACE	 DEPLOYABLE	 ERECTABLE

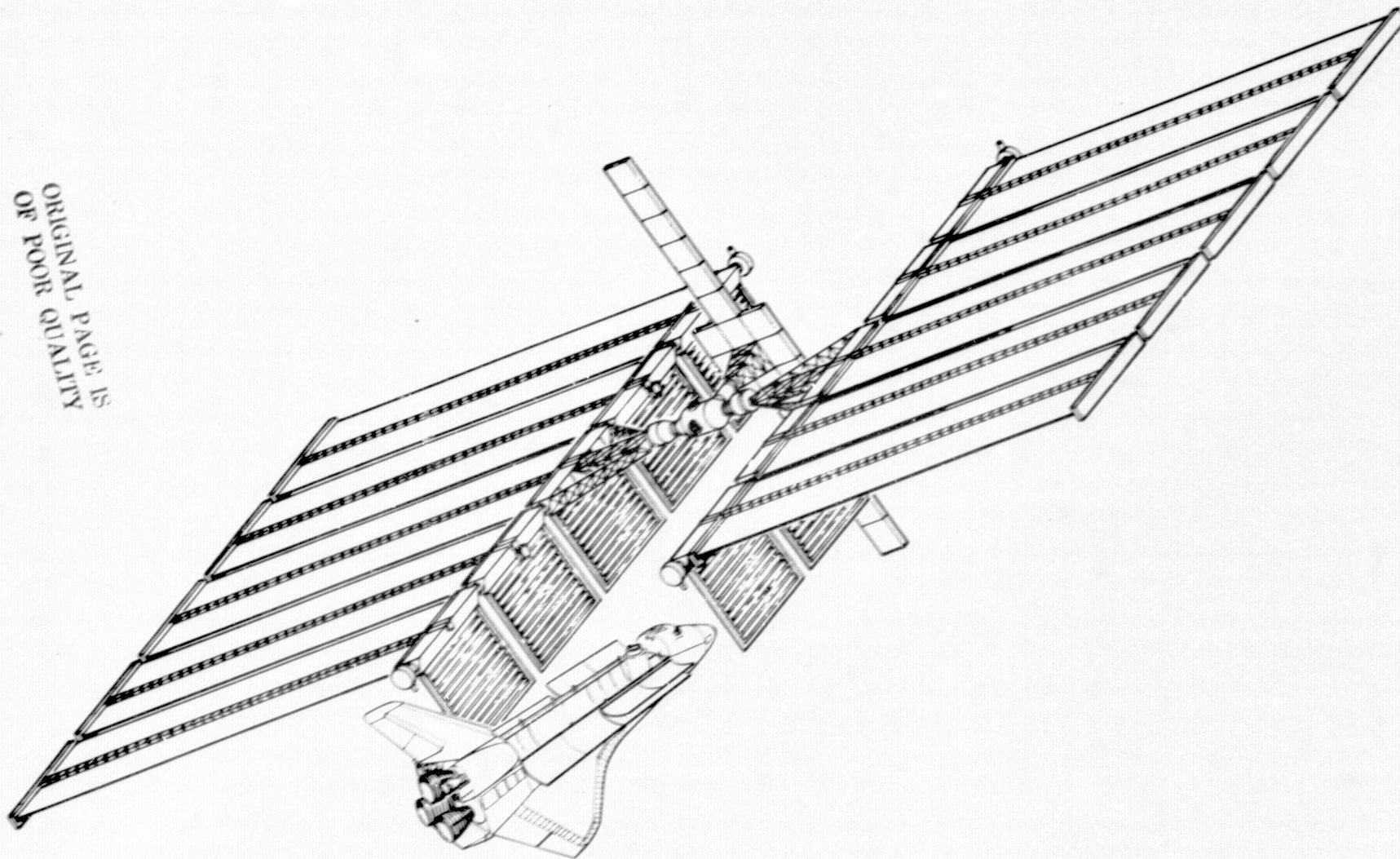
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SOLAR POWER DEMONSTRATION TEST ARTICLE (ONE VERSION)

A logical extension of SEP/PEP/LPM power module development can result in the solar power demonstration test article shown on the facing chart. This SPS microwave transmission test system would generate 550 kW (full sun) and can be put into low-earth orbit with five Shuttle flights. A multiple docking port provides Shuttle tended operation. The power conversion system consists of ten standard SEP/PEP deployable silicon cell arrays. The microwave antenna utilizes erectable fold-out deployment techniques. The vehicle can be disassembled and returned to earth. Ion thrusters using argon fuel provide stabilization and control. The test platform is structurally capable of orbital transfer to geostationary orbit utilizing solar electric propulsion.

SOLAR POWER DEMONSTRATION TEST ARTICLE

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SPS TEST PLATFORM COMPARATIVE CHARACTERISTICS

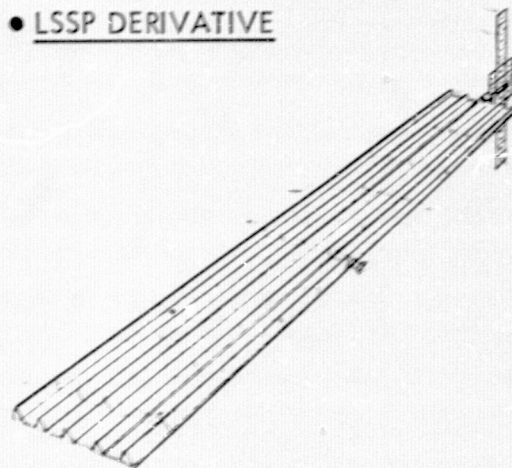
The facing chart summarizes the principal system characteristics that could be achievable for LSSP and SEP/PEP derivatives of an SPS development test article. An SPS test article evolving out of the large space structures program with beam machine construction is likely to provide the highest degree of SPS technology verification for subsystems other than the MPTS.

If the SPS system is baselined for GaAs solar cells the LSSP derivative again provides the best technology verification results, providing that an intensive GaAs solar array development program is initiated by FY 1980.

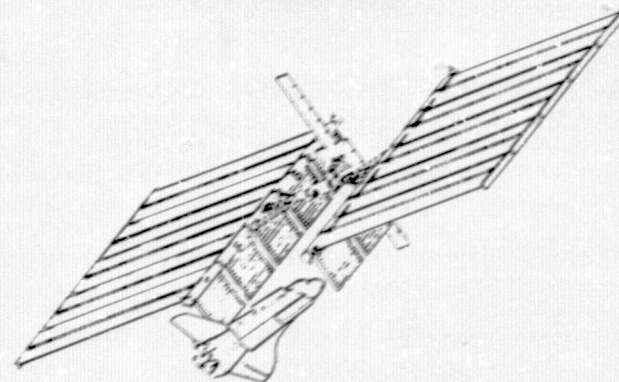
SPS TEST PLATFORM COMPARATIVE CHARACTERISTICS

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• LSSP DERIVATIVE



• SEP/PEP DERIVATIVE



• POWER CONVERSION

✓ COULD BE ALL GaAs OR HYBRID

✓ SEP SILICON ROLL-OUT ARRAY -
COULD BE HYBRID

• POWER DISTRIBUTION

✓ DESIGN OPTIMIZED FOR LEO &
GEO HI-VOLTAGE OPERATION

✓ OPTIMIZED FOR LEO OPERATION
≈ 200 VOLTS

• STRUCTURE

✓ PROTOTYPICAL BEAM MACHINE
FABRICATION/JOINING

✓ DEPLOYABLE ARRAYS - ERECTABLE
ANTENNA

• POWER TRANSMISSION

✓ MEDIUM POWER AMPLIFIERS

✓ MEDIUM POWER AMPLIFIERS

• DEGREE OF SPS VERIFICATION

✓ HIGHEST

✓ MODERATE

• PROBABILITY OF EARLY IMPLEMENTATION

✓ MODERATE

✓ HIGHEST

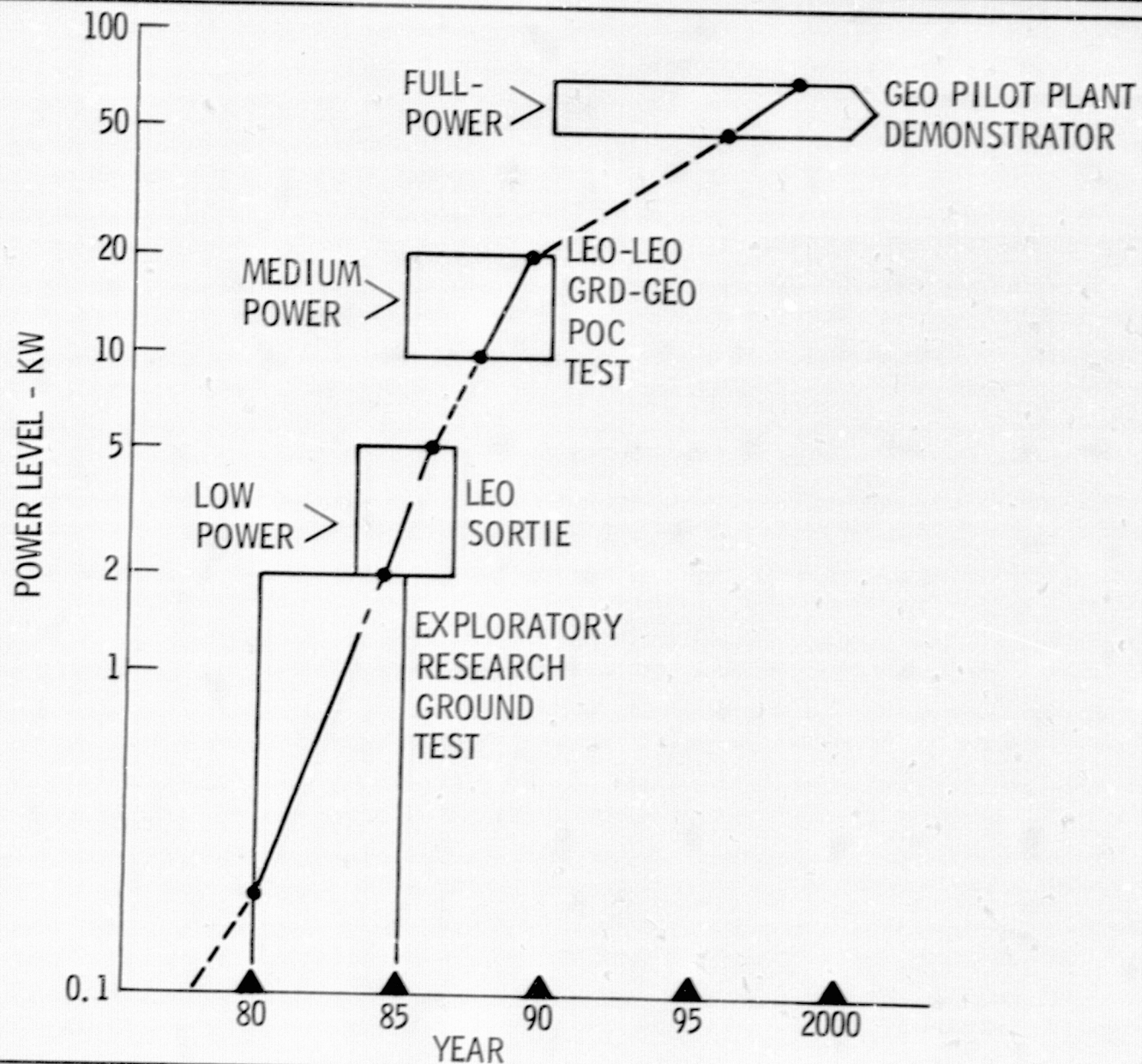


SPS POWER AMPLIFIER POWER LEVEL PROJECTION

SPS klystron development is the critical technical path that will determine the design and performance characteristics of the SPS development test platform microwave antenna. The technical and funding depth of the contemplated microwave exploratory research program will establish the power levels of flight rated power amplifiers as shown on the facing chart.

Power levels from 5 to 20 kW span the probable klystron performance range available for SPS orbital microwave testing in the late 80's.

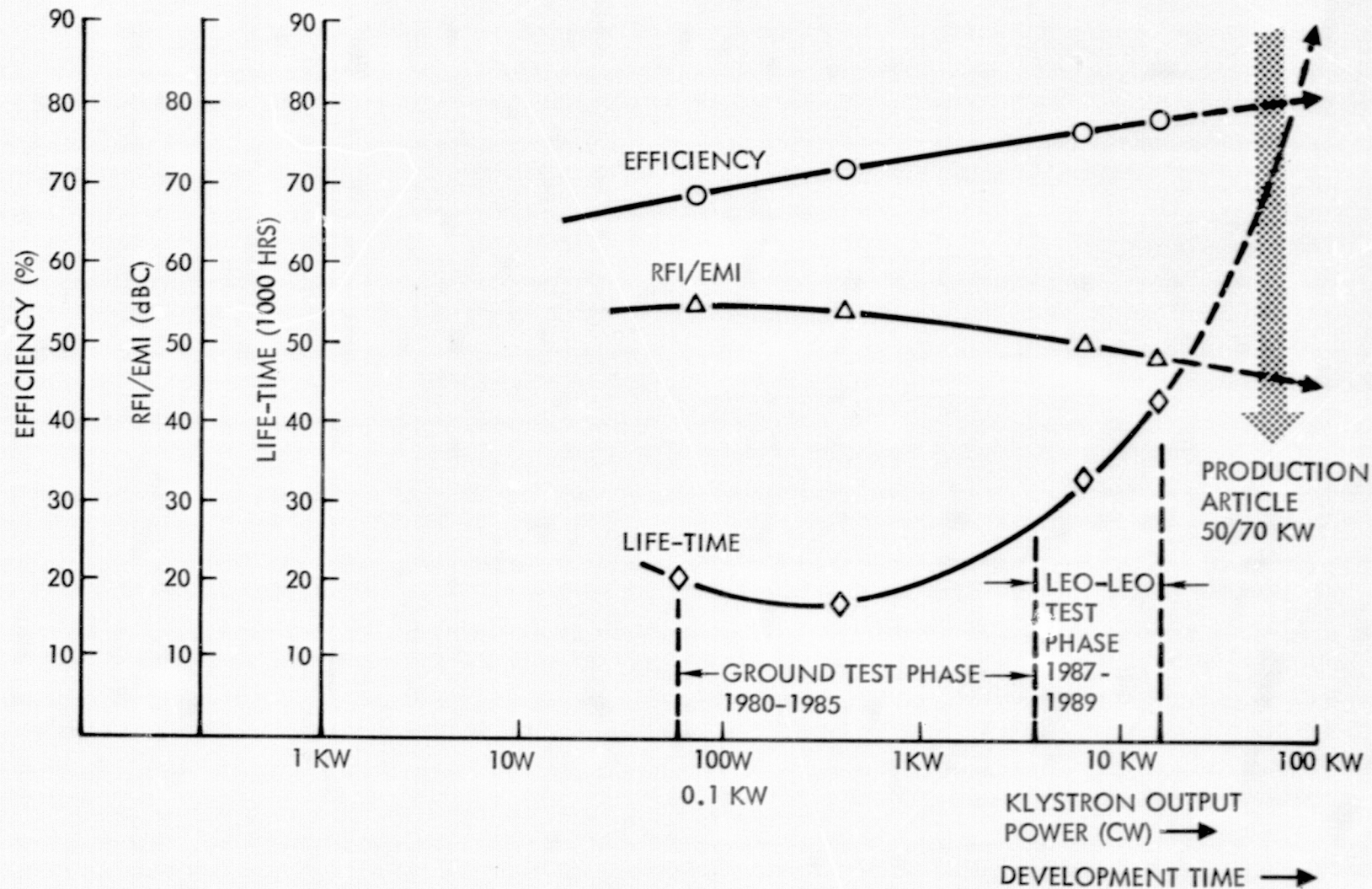
SPS POWER AMPLIFIER POWER LEVEL VS DEV. TIME



KLYSTRON PERFORMANCE PROJECTION TRENDS

A progressive development build-up of SPS power amplifier performance capability as a function of development time and power level is summarized on this chart. The ground test phase will involve low power amplifiers in the range of 1/10 kW to 5 kW. Orbital and ground to GEO linear array antennas will be on the order of 5 to 20 kW with progressive improvement in tube efficiency, MTBF and RFI suppression.

PERFORMANCE PROJECTION TRENDS BASED ON GROUND TEST AND LEO TEST OF VARIOUS KLYSTRON OUTPUT POWER LEVELS



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TECHNOLOGY VERIFICATION OVERVIEW

This chart is self-explanatory.

TECHNOLOGY VERIFICATION OVERVIEW

GOAL: To project the feasibility and economic advantages of the SPS system using lower cost, power and size prototype elements. By extrapolating the data obtained with the lower power elements, the performance of the final SPS system configuration can be estimated with minimum uncertainty.

OBJECTIVE:

Near Term: To obtain sufficient data with lower power klystron and solid state transistor elements during the 1980-1985 time frame to provide performance projections on the SPS MPTS power module life-time, efficiency and RFI/EMI potential with minimum uncertainty.

Long Term: To develop enough of a prototype SPS system (in low earth orbit) through 1987 in order to establish the feasibility and economic advantages of the SPS system.

APPROACH:

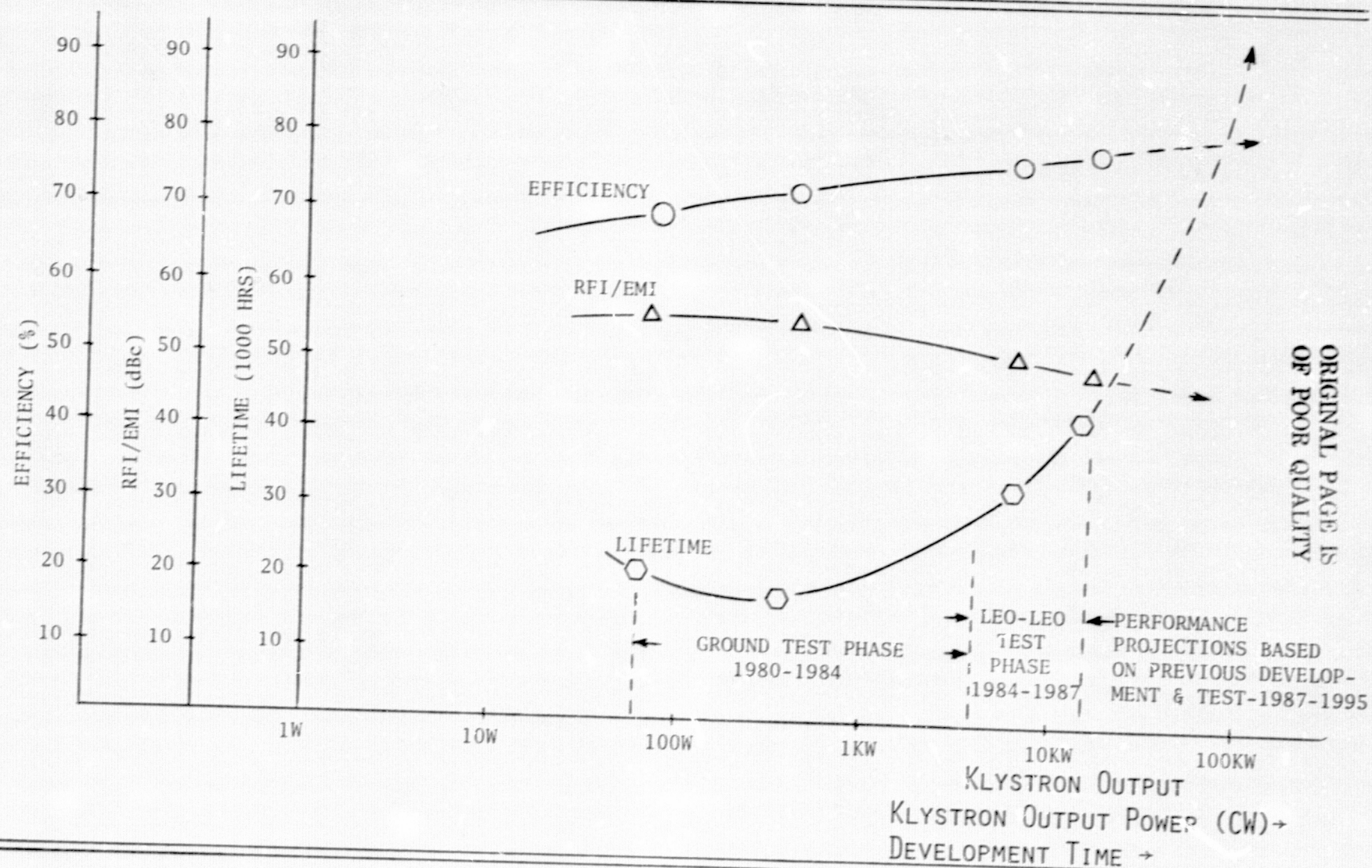
Low power (up to 2KW) elements will be ground tested in order to evaluate safety problems and procedures for test and maintenance at LEO. Intermediate power (up to 15KW) elements will be tested at LEO in order to determine reasonable expected performance levels for the final SPS system configuration.



PERFORMANCE PROJECTION TRENDS BASED ON GROUND TEST AND
LEO TEST OF VARIOUS KLYSTRON OUTPUT POWER LEVELS

A progressive development build-up of SPS power amplifier performance capability as a function of development time and power level is summarized on this chart. The ground test phase will involve low power amplifiers in the range of 1/10 kW to 5 kW. Orbital and ground to GEO linear array antennas will be on the order of 5 to 20 kW with progressive improvement in tube efficiency, MTBF and RFI suppression.

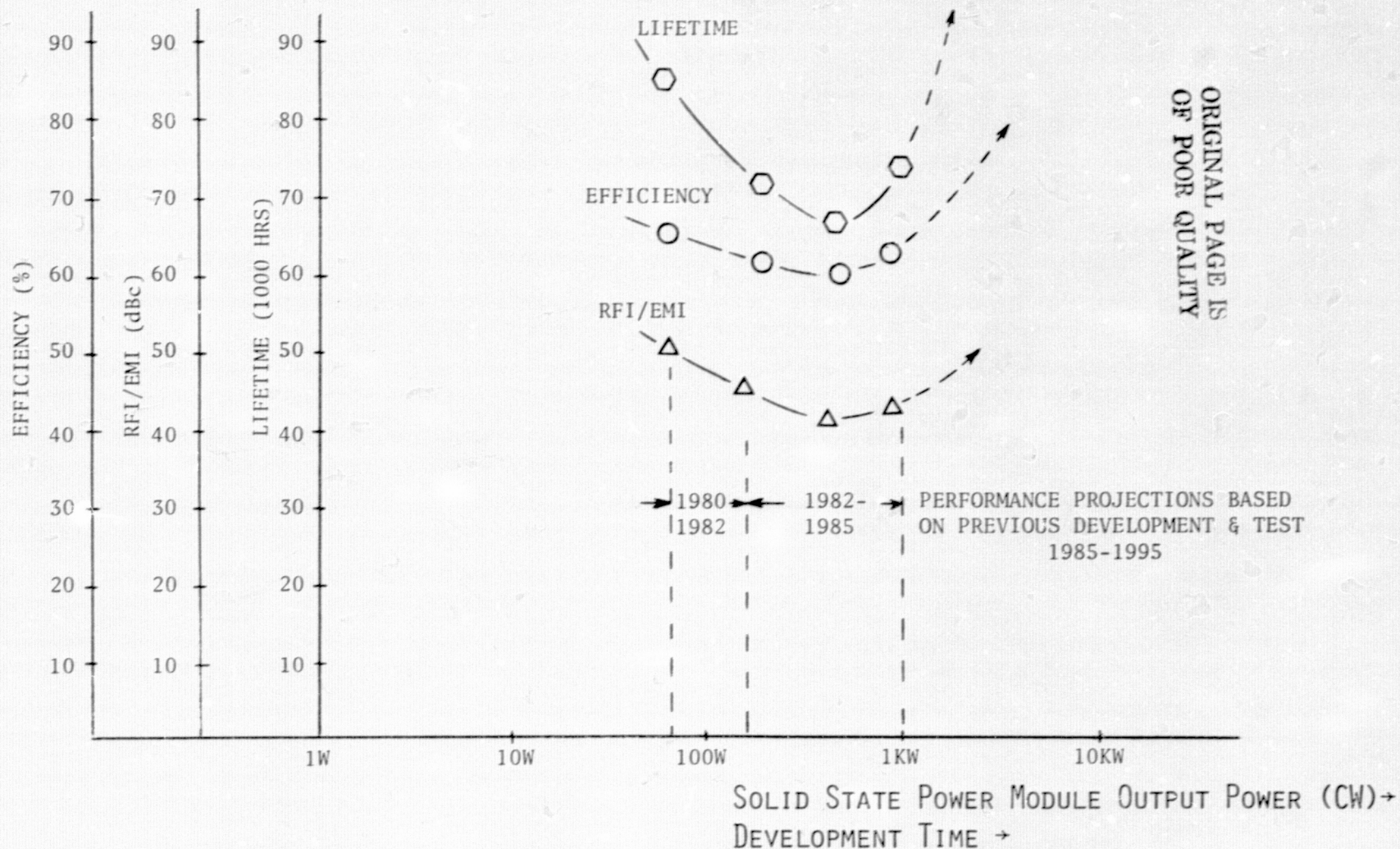
PERFORMANCE PROJECTION BASED ON GROUND TEST AND LEO TEST OF VARIOUS KLYSTRON OUTPUT POWER LEVELS



PERFORMANCE PROJECTION BASED ON GROUND TEST AND LEO TEST
OF VARIOUS SOLID-STATE DEVICE OUTPUT POWER LEVELS

This chart illustrates expected device improvements, as a function of time, for solid-state devices.

PERFORMANCE PROJECTION BASED ON GROUND TEST AND LEO TEST OF VARIOUS SOLID STATE DEVICE OUTPUT POWER LEVELS



SPS MPTS GROUND TEST METHODOLOGY

This chart is self-explanatory.

SPS MPTS GROUND TEST METHODOLOGY

- A SCALED DOWN VERSION OF THE SPS MICROWAVE ANTENNA ARRAY CAN BE EVALUATED TO DETERMINE THE EFFECTS OF VARIOUS COMPONENTS (THESE GROUND TEST RESULTS CAN BE INTEGRATED WITH LEO-LEO TEST RESULTS TO PRODUCE A FINAL PERFORMANCE PROJECTION FOR THE FULL SCALE SYSTEM).
- USE GROUND TEST RESULTS TO FORM PRELIMINARY PERFORMANCE PROJECTIONS FOR THE FULL SCALE SPS MPTS ARRAY.
- GROUND TEST MAINTAINABILITY EVALUATIONS (COMBINED LATER WITH LEO-LEO MAINTAINABILITY EVALUATIONS) COULD BE SIGNIFICANT DESIGN DRIVERS FOR THE KLYSTRON.
- LIFE TESTING WOULD REQUIRE CONTINUOUS MONITORING OF CW OUTPUT, SPURS, HARMONICS, NOISE AND LEAKAGE UNDER ACCELERATED OPERATING CONDITIONS.

SPS MPTS GROUND TEST PERFORMANCE EVALUATION

The following two charts describe the various factors that must be measured and/or evaluated to determine subarray performance characteristics. The actual tests would utilize scaled down subarray structures where appropriate.

SPS MPTS GROUND TEST PERFORMANCE EVALUATION (USING SCALED DOWN SUB-ARRAY STRUCTURES)

1. MICROWAVE POWER TRANSMISSION EFFICIENCY

- MEASURE EFFICIENCY (USING RECEIVING ANTENNA PROBE) OVER VARIOUS THERMAL CONDITIONS AND VARIOUS LEVELS OF INDUCED PHASE ERROR.
- EVALUATE MODULE AND SUB-ARRAY SIDE LOBES AND BACK LOBES. ANALYZE DEVIATIONS FROM THEORETICAL.

2. MICROWAVE POWER TRANSMISSION - THERMAL EFFECTS

- EFFECTS OF THERMAL STRESS ON SUB-ARRAY STRUCTURE AND RADIATION PATTERN.
- EFFECTS OF THERMAL STRESS ON KLYSTRON AND SOLID STATE DEVICE LIFETIMES.

3. ECLECTIC RETRODIRECTIVE PHASE CONTROL SYSTEM EVALUATION

- DETERMINE DETECTABILITY OF PILOT PHASE CONTROL SIGNAL-IN HIGH INTERFERENCE ENVIRONMENT.
- EVALUATE PERFORMANCE AND LIFE-TIME OF PHASE CONTROL ELECTRONICS AT VARIOUS THERMAL LEVELS.
- DETERMINE TOLERANCE LIMITS OF THE PHASE CONTROL SYSTEM.

4. MICROWAVE POWER TRANSMISSION SYSTEM EMI/RFI STUDIES

- MEASURE LEAKAGE, SPURIOUS AND NOISE OUTPUTS OF KLYSTRON AND SOLID STATE POWER MODULES AND SUB-ARRAYS.
- MONITOR RFI GENERATED BY INTERCONNECTS AND POWER SUPPLIES.
- DETERMINE SIDE LOBE LEVEL VARIATIONS WITH VARIOUS MODULES SHUT DOWN.



SPS MPTS GROUND TEST PERFORMANCE EVALUATION
(USING SCALED DOWN SUB-ARRAY STRUCTURES) CON'D

- KLYSTRON X-RAY EMISSION STUDIES

5. MAINTAINABILITY EVALUATION

- DETERMINE EFFECTS OF TUBE REPLACEMENT ON SUB-ARRAY PERFORMANCE AND EVALUATE REPLACEMENT TOLERANCE LIMITATIONS AND REQUIREMENTS.
- SAFETY ASPECTS AND AUTOMATED TECHNIQUES.



PLANS FOR DESIGN, DEVELOPMENT
AND TEST OF CRITICAL SPS MPTS COMPONENTS

Satellite Systems Division
Space Systems Group



Rockwell
International

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TECHNOLOGY VERIFICATION
GROUND TEST OF BASIC ARRAY ELEMENTS

This chart is self-explanatory.

TECHNOLOGY VERIFICATION

GROUND TEST OF BASIC ARRAY ELEMENTS

- KLYSTRON AND KLYSTRON POWER SUPPLY PERFORMANCE

TEST CURRENTLY AVAILABLE LOW POWER KLYSTRONS IN THE BASIC POWER MODULE CONFIGURATION AT 70W, 200W AND 1KW OUTPUT LEVELS TO OBTAIN DATA ON PERFORMANCE, LIFETIME AND TEST PROCEDURE POLICY. COMBINING THIS DATA WITH THE MEDIUM POWER TUBES DEVELOPED FOR THE LEO-LEO TEST, PROJECTED PERFORMANCE LEVELS AND LIFETIMES FOR THE HIGH POWER MPTS TUBE CAN BE EXTRAPOLATED ON A LINEAR OR NON-LINEAR CURVE OR TREND LINE.

- SUB-ARRAY PERFORMANCE

SEVERAL 3x3 METER OR 5x5 METER ARRAYS CAN BE CONSTRUCTED. MEASUREMENTS WOULD BE TAKEN ON PERFORMANCE, THERMAL PROFILE AND PHASE NOISE/PHASE ERROR PROBLEMS CAUSED BY VARIOUS INDIVIDUAL KLYSTRONS, INTERACTIONS AND DEFFICIENCIES IN POWER SUPPLY GROUNDING. ESTIMATES COULD ALSO BE OBTAINED ON LIFETIME AND MAINTAINABILITY REQUIREMENTS FOR THE HIGHER POWER TUBES BY DATA OBTAINED FROM THE LOWER POWER SUB-ARRAY PERFORMANCE.

- PHASE CONTROL SYSTEM PERFORMANCE

ANALYZE DETECTION PROBLEMS OF SIMULATED RETRODIRECTIVE PILOT SIGNAL IMMERSSED IN STRONG INTERFERENCE SIGNALS AND BACKGROUND NOISE LEVELS.

GROUND TEST ANTENNA HARDWARE REQUIRED FOR
TECHNOLOGY VERIFICATION

This chart indicates initial make/buy decisions for the major elements of the antenna test article.

GROUND TEST ANTENNA HARDWARE REQUIRED FOR TECHNOLOGY VERIFICATION

<u>GROUND FACILITY PARTS</u>	<u>DEVELOP</u>	<u>BUY</u>
● MULTIPLE PHASE CONJUGATING CIRCUITS	X	
● PHASE REFERENCE DISTRIBUTION SYSTEM	X	
● MULTIPLE DC-RF CONVERTERS	X	
● MULTIPLE PILOT RECEIVE CIRCUITS	X	
● DIPLEXING FILTERS	X	
● WAVEGUIDE		X
● DRIVER AMPLIFIERS	X	
● POWER MANAGEMENT HARDWARE (CONVERTERS, SWITCHES, REGULATORS)		X
● DIRECTIONAL COUPLERS		X



TECHNOLOGY VERIFICATION TEST ITEMS

The next two charts present a preliminary list of tests that must be made to characterize the operating power conversion devices and also indicates applicability to the klystron and solid-state amplifier approaches.

TECHNOLOGY VERIFICATION TEST ITEMS (NEAR TERM)*

TEST ITEM	KLYSTRON	SOLID STATE
1. POWER SUPPLY EMI		
- POWER LEAD EMISSIONS	X	X
- INCIDENTAL COUPLING	X	X
- FM & PM NOISE OVER FREQUENCY	X	X
2. SPURIOUS OUTPUTS	X	X
3. AM/PM CONVERSION	X	X
4. THIRD ORDER INTERMODULATION PRODUCTS	X	X
5. HARMONICS	X	X
6. VOLTAGE REGULATION REQUIREMENTS AND EFFECTS	X	X
7. RFI SUSCEPTIBILITY	X	X
8. MAGNETIC FIELD SUSCEPTIBILITY	X	
9. RADIATION SUSCEPTIBILITY	X	X
10. SATURATION DRIVE VS. PHASE SHIFT	X	X
11. MODULE RADIATION PATTERN (INCLUDES GRATING LOBES AND SIDE LOBES)	X	X
12. THERMAL PROFILE	X	X
13. EFFICIENCY	X	X
14. DRIVE LEVELS	X	X
15. MTBF ESTIMATES (DEVICE & POWER SUPPLY)	X	X
16. X-RAY EMISSION	X	
17. SATURATION GAIN	X	X
18. GAIN STABILITY	X	X



TECHNOLOGY VERIFICATION TEST ITEMS (NEAR TERM)* CONT'D

TEST ITEM	KLYSTRON	SOLID STATE
19. GAIN SLOPE	X	X
20. WARM UP CHARACTERISTICS	X	X
21. EFFECTS ON PHASE CONTROL CIRCUITRY	X	X
22. TRANSIENTS @ START-UP AND SHUT-DOWN	X	X
23. POWER SUPPLY BREAK-DOWN	X	

* LONG TERM TEST REQUIREMENTS WILL BE DRIVEN BY THE NEAR-TERM GROUND TEST RESULTS.



PREPARATION OF INTEGRATED GROUND TEST RANGE

This chart identifies several equipments and procedures that are applicable to the use of an integrated ground based test range for the evaluation and checkout of the elements of the microwave antenna system.

PREPARATION OF INTEGRATED GROUND TEST RANGE

- ANECHOIC CHAMBER PREPARATION (POWER MODULE AND 3 x 3 METER SUBARRAY RADIATION PATTERNS).
- KLYSTRON/SOLID STATE DEVICE TEST CHAMBERS AND TEST STANDS (WITH POWER SUPPLY AND ELECTRONIC TEST EQUIPMENT RACKS).
- POWER MODULE ASSEMBLY AND COMPONENT TEST AREA (WITH SEPARATE POWER SUPPLY AND ELECTRONIC TEST EQUIPMENT RACKS).
 - 1. KLYSTRON AREA
 - 2. SOLID STATE DEVICE AREA
- TEST EQUIPMENT CALIBRATION AND STORAGE AREA.
- TEST PROCEDURE DATA CONTROL AND PROCEDURE EVALUATION AREA (ALSO ESTABLISHES DATA/TEST PROCEDURE INTEGRATION WITH LEO-LEO TEST PHASE).
- GROUND TEST RANGE SAFETY EVALUATION INTEGRATED WITH LEO-LEO TEST PHASE.



KLYSTRON POWER MODULE PROTOTYPE DEVELOPMENT AND GROUND TEST

This chart lists a few of the major activities needed to fabricate and test development and prototype klystron power conversion elements of the MPTS.

KLYSTRON POWER MODULE PROTOTYPE DEVELOPMENT AND GROUND TEST

- PURCHASE FORTY LOW POWER 2.5 GHZ KLYSTRONS (WITH POWER SUPPLIES) WITH DEPRESSED COLLECTORS FOR INDIVIDUAL TEST AND EVALUATION.
- SET UP TEN OF THE KLYSTRONS FOR ACCELERATED LIFE TEST.
- DEVELOP PROTOTYPE PHASE CONTROL ELECTRONICS.
- PURCHASE TWO VKS-773 50KW, 75% EFFICIENT, 28KV WATER COOLED 2.5 GHZ KLYSTRONS WITH CASCADED PRE-BUNCHERS FOR TESTING EMI (HARMONICS, INTERFERENCE, SPURIOUS, PHASE NOISE, ETC) AND ESTABLISHING PRELIMINARY HIGH POWER TEST PROCEDURE REQUIREMENTS.
- FROM PRELIMINARY GROUND TEST DATA, INITIATE DEVELOPMENT OF MEDIUM POWER LEO-LEO KLYSTRONS WITH DEPRESSED COLLECTORS.
- ASSEMBLE 30 KLYSTRON POWER MODULES. OBTAIN INDIVIDUAL PATTERNS THERMAL PROFILES AND TEST DATA.
- INTEGRATE 6 POWER MODULES ON 3 x 3 METER SUBARRAYS. OBTAIN INDIVIDUAL PATTERNS THERMAL PROFILES AND TEST DATA FOR EACH SUBARRAY.
- INTEGRATE ALL FIVE SUBARRAYS AND PHASE CONTROL ELECTRONICS INTO AN ANTENNA ARRAY AND OBTAIN INDIVIDUAL PATTERNS, THERMAL PROFILES AND TEST DATA FOR THE ARRAY.
- EXTRAPOLATE RESULTS TO PREDICT 1KM SPS ARRAY PERFORMANCE AND INTEGRATE WITH LEO-LEO TEST PHASE.



SOLID-STATE POWER MODULE DEVELOPMENT AND GROUND TEST

This chart lists some of the major activities associated with the development of a solid-state based version of the MPTS power module.

SOLID STATE POWER MODULE DEVELOPMENT AND GROUND TEST

- SEMICONDUCTOR SUBSTRATE, PROCESS, PACKAGING AND THERMAL PROFILE EVALUATION.
- DECISION ON TRANSISTOR CANDIDATE.
- DECISION ON CLASS-C OR CLASS-E AMPLIFIER CONFIGURATION.
- LSI MANUFACTURING PROCESS EVALUATION (PRODUCTION PROCESSING METHODOLOGY IMPROVEMENTS, LEVEL-OF-DIFFICULTY AND FEASIBILITY EVALUATIONS) DONE AT NASA MSFC SOLID STATE CONTROLLED PROCESSING AREA. PARALLEL EFFORT AT ROCKWELL SCIENCE CENTER AND ELECTRONIC RESEARCH CENTER ON FEASIBILITY EVALUATIONS, MATERIAL INTERFACE PROBLEMS AND PRODUCTION COST ASSESSMENTS.
- DEVELOP 1KW SOLID STATE POWER MODULE DATA OBTAINED INCLUDES PERFORMANCE RADIATION PATTERNS AND THERMAL PROFILE.
- DEVELOP FIVE 1x1KW SOLID STATE POWER MODULES AND ASSEMBLE INTO A 3 x 3 METER SUBARRAY. OBTAIN RADIATION PATTERN THERMAL PROFILE AND TEST DATA FOR THE SOLID STATE SUBARRAY.
- CONTINUE DEVELOPMENT OF A HIGHER POWER (UP TO 5KW) SOLID STATE POWER MODULE.
- EXTRAPOLATE RESULTS TO PREDICT SPS PERFORMANCE USING SOLID STATE DEVICES. PREPARE FOR CHOICE BETWEEN KLYSTRONS AND SOLID STATE.



DESCRIPTION OF NASA MSFC CONTROLLED PROCESSING AREA (CPA)

This chart is self-explanatory.

DESCRIPTION OF NASA MSFC CONTROLLED PROCESSING AREA (CPA) FOR HIGH DENSITY CIRCUIT TECHNOLOGY

- THE CONTROLLED PROCESSING AREA (CPA) FOR HIGH DENSITY CIRCUIT TECHNOLOGY IS PRESENTLY BEING USED IN THE DEMONSTRATION OF NEW STATE-OF-THE-ART LARGE SCALE INTEGRATED CIRCUIT (LSIC) PROCESSING EQUIPMENT DEVELOPED FOR NASA.
- THE CPA WILL FUNCTION AS A PRECISELY CONTROLLED EXPERIMENTAL SOLID STATE INTEGRATED CIRCUIT DEVELOPMENT FACILITY AND WILL FABRICATE THESE CIRCUITS IN A HANDS-OFF MANNER.
- A FOLLOW-ON STEP IS THE DEVELOPMENT AND DEMONSTRATION OF TOTAL AUTOMATED TECHNIQUES WHICH PROVIDE COST EFFECTIVE METHODS TO PRODUCE UNIFORM PRODUCTS.
- LOCATED NEARBY ARE SUPPORTING AREAS WHICH GENERATE THE PHOTOLITHOGRAPHIC PROCESS MASKS AND BOND AND PACKAGE THE LSIC CHIPS.
- CPA STATUS:

CURRENT CONDITION	90% COMPLETE
ACTIVATION DATE	1978
REPLACEMENT COST	\$2M (1978)



TECHNICAL SPECIFICATIONS FOR NASA MSFC LSIC CONTROLLED
PROCESSING AREA

This chart is self-explanatory.

TECHNICAL SPECIFICATIONS FOR NASA MSFC LSIC CONTROLLED PROCESSING AREA

- FACILITIES:

STANDARD WORKING AREA: 1700 SQ. FT.
CLEAN ROOM AREA: 1600 SQ. FT.

- CLEAN ROOM SPECIFICATIONS:

PHOTOLITHOGRAPHY: 16'x20' - 100 CLASS
ETCH VATION: 20'x20' - 10,000 CLASS
FURNACE ROOM: 43'x20' - 10,000 CLASS
ENTRY SIZE: 72"x80" POSITIVE PRESSURE
FLOOR TYPE: FALSE FLOOR
TEMPERATURE CONTROL: $70 \pm 2^{\circ}\text{F}$
HUMIDITY CONTROL: $42 \pm 3\%$
VACUUM: BUILTIN SYSTEM
AIR CONDITION: 40 TONS

- UTILITIES:

GASEOUS AND LIQUID NITROGEN
SUPER CLEAN GRADE COMPRESSED AIR
DISTILLED AND DEIONIZED WATER
DEIONIZED WATER REPOLISHING PLANT
POWER DISTRIBUTION ROOM
FURNACE FLUE GAS CONTROL: POSITIVE EXHAUST, FLUE GAS SCRUBBERS WITH GAS FLOW MONITORS.

- EQUIPMENT:

VACUUM METALIZATION SYSTEM
METAL PLATING SYSTEM
OXIDATION-DIFFUSION FURNACE
EPITAXIAL REACTOR
CHEMICAL VAPOR DEPOSITION FURNACE
ION IMPLANATION SYSTEM
PHOTOLITHOGRAPHY SYSTEM
MASK ALIGNMENT SYSTEM
SPUTTERING SYSTEM
WAFER TRANSPORT MECHANISM (TRAM)
GAS TRACK
TRAM WAFER TRAFFIC CONTROL SYSTEM

CAPABILITIES OF NASA MSFC CONTROLLED PROCESSING AREA

This chart is self-explanatory.

CAPABILITIES OF NASA MSFC CONTROLLED PROCESSING AREA
FOR HIGH DENSITY CIRCUIT TECHNOLOGY

- THE CPA SERVES AS THE ULTIMATE TECHNOLOGY VERIFICATION TOOL FOR DESIGN, DEVELOPMENT AND EVALUATION OF NEW STATE-OF-THE-ART LARGE SCALE INTEGRATED CIRCUIT (LSIC) PROCESSING AND PRODUCTION TECHNIQUES REQUIRED BY HIGH TECHNOLOGY PROGRAMS SUCH AS SPS.
- THE NASA MSFC CPA PROVIDES THE CAPABILITY TO SUPPORT TECHNOLOGY FEASIBILITY EVALUATIONS, INVESTIGATE PRODUCTION PROCESSING METHODOLOGY IMPROVEMENTS AND DETERMINE PRODUCTION COSTS AND LEVEL-OF-DIFFICULTY ASSESSMENTS FOR EARLY PHASES OF THE SPS TEST PROGRAM.
- THE SYSTEM IS PRESENTLY SET UP FOR SILICON DEVICE PROCESSING. THE SYSTEM CAN ALSO BE ADAPTED TO GALLIUM ARSENIDE TECHNOLOGY FOR EVALUATING SPS SOLAR CELL, RECTENNA DIODE AND TRANSISTOR DEVICE CANDIDATES AND ASSOCIATED PROCESSING TECHNIQUES.



ADAPTABILITY OF NASA MSFC CONTROLLED PROCESSING AREA
FOR SOLAR POWER SATELLITE

This chart is self-explanatory.

ADAPTABILITY OF NASA MSFC CONTROLLED
PROCESSING AREA FOR SOLAR POWER SATELLITE
DEVICE AND PROCESSING REQUIREMENTS

TECHNOLOGY/DEVICE END PRODUCT	BASIC PRODUCT	CARRIER OR PURGE GAS	REACTANTS	TEMPERATURE RANGE	DOPANTS
• CONVENTIONAL SILICON MOS LSI TECHNOLOGY*	EPITAXIAL SILICON DEPOSITION ON SILICON	H ₂	SiH ₄	1100°C-1150°C	PH ₃ , B, H ₂ , H ₄
• HIGH VOLUME GAAs SOLAR CELL TECH.	EPITAXIAL GAAs DEPOSITION ON Al ₂ O ₃	H ₂	(CH ₃) ₃ GA & AsH ₃	655°C-700°C	AsH ₃ -
• HIGH VOLUME GAAs TRANSISTOR OR DIODE TECHNOLOGY	EPITAXIAL GAAs DEPOSITION ON GAAs, Al ₂ O ₃ OR SPINEL	H ₂	GAAs-GA-TE MELT. ♦	700°C	TE
		H ₂	(CH ₃) ₃ GA & AsH ₃	630°C-720°C	Cu, Zn Fe, S
* PRESENT SET-UP ♦ LIQUID PHASE EPITAXY. ALL OTHERS ARE CHEMICAL VAPOR DEPOSITION.					

